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MARTIN MARIETTA

HYBRID PROPULSION TECHNOLOGY PROGRAM

Phase I—Final Report

Volume II Technical Discussion

Contract NAS8-37775

23 October 1989

NASA
National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

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Volume II Technical Discussion

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Prepared for: National Aeronautics and Space Administration
George C. Marshall Space Flight Center
Huntsville, AL 35812

Prepared by: Aerojet Solid Propulsion
Aerojet TechSystems
Martin Marietta Aerospace

c/o Aerojet Solid Propulsion
Box 15699C
Sacramento, CA 95813-1699

VOLUME II TECHNICAL DISCUSSION

Table of Contents

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TECHNICAL DISCUSSION

In Task 1, Concept Definitizing, We:

- Developed Hybrid Propulsion Concepts, Based Upon NASA Requirements, Our Past Experience and Data Base, and Using Our Concept Generation Matrix Approach
- Developed Hybrid Propulsion Screening Criteria, Based Upon NASA Priorities and Our Experience
- Yes/No Screen All Concepts With the Criteria. A Single "No" Eliminated Any Candidate
- Defined Acceptable Concepts in All Categories and Subcategories, Which We Call Levels 1, 2, and 3

The Following Charts in This Section Flow in Reverse Order to Show Results First and Bases Last

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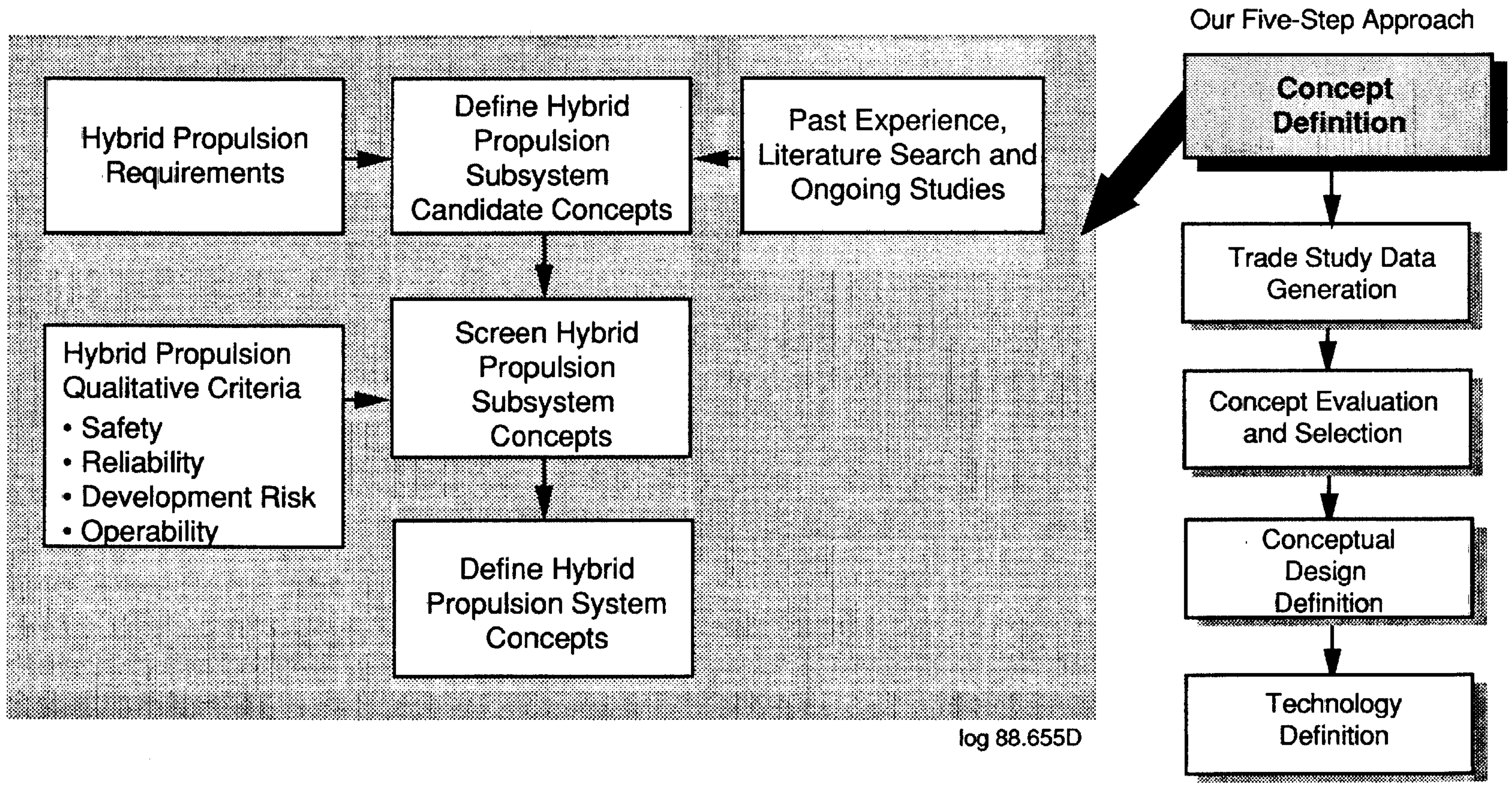
OUR CONCEPT DEFINITION AND RANKING PROCESS IS THOROUGH, LOGICAL, AND RESPONSIVE TO THE SOW



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DEFINE HYBRID PROPULSION SYSTEM CONCEPTS

First and Most Fundamentally, We Must Have Propellants to Burn and Know Why We Have Them. Secondly, We Must Know How to Best Combust Them and Why. Finally, We Need to Investigate All of the Major Subsystems Needed to Support the Combustion, Such as Feed Systems, Systems, Tank Pressurization, Cooling, etc.

HRB STUDY CONCEPT GENERATION AND SCREENING LEVELS

- Level 1 — Propellant (Physical and Chemical Options)
- Level 2 — Combustion (Methods and Solid Grain Chemistry)
- Level 3 — Subsystem Concepts

$\boxed{\text{Concepts Generated and Screened by Level in Series}}$

We Screened to Three Liquid Oxidizers and Ten Solid Fuels at Level 1, LO_2 and Two Densified LO_2s. We Screened to Four Combustion Schemes at Level 2. Three Have Some Oxidizer in the Solid Fuel Grain; the Conventional Hybrid Does Not. The First Two Have Oxidizer Injection Into the Solid Case Only, the SLSC Has Injection Aft of the Solid Case Into a Thrust Chamber, and the Last One Has Both. These Are Not Selected, but They Are Acceptable Options. Details, Reasons, and Justification of the Screening Are Shown Later in This Section

LEVELS 1 AND 2 SCREENING RESULTS

Level 1 — Propellants

Liquid Oxidizer

- LO_2
- $LO_2 + N_2O_4(S)$
- $LO_2 + H_2O_2(S)$

Solid Fuel

- Ten Formulations

Level 2 — Combustion

- Conventional Hybrid
- Quasi-Hybrid
- Solid-Liquid Staged Combustion
- Quasi-Hybrid-Liquid Staged Combustion

These Next Two Sheets Show the 15 Subsystems Studied, the Options Considered, and the Priority Assigned to Each. Priorities Prevented Overrunning This Important Activity

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LEVEL 3 HRB SUBSYSTEM OPTIONS

	Subject	Option	Priority* Assigned
1.	Liquid Propellant Tank Pressurization	• Pressure Fed • Pump Fed - Autogenous Tank Pressurization - Other: He, Azide, etc.	A (A=Mandatory, B=Secondary Importance, C=As Req'd)
2.	Turbopump-Fed Drive	• Topping Cycle • Bleed Cycle	
3.	Type TCA	• DeLaval • Plug Cluster • Plug • ED	A A
4.	Number of TCAs	• One Chamber/One Nozzle • Four Chambers/Four Nozzles	 A
5.	Solid Propellant Case	• Metallic • Cylindrical • Composite • Tapered	 A
6.	Solid Grain Design	• Multiperforate • Internal Burn	
7.	Liquid Propellant Tank	• Metallic • Cylindrical • Composite • Tapered • Overwrapped Metallic	B B
8.	SLSC Liquid Feed System	• Pressure Lines • External Lines • Suction Lines • Internal Lines	 C
9.	Active Thrust, I_{sp}, PU Control System	• Liquid Only • Liq/Solid	 A

*Basis: Enable Screening Activities

LEVEL 3 HRB SUBSYSTEM OPTIONS (Cont)

	Subject	Option	Priority* Assigned
10.	Thrust Vector Control	• Injection: Gas, Liq. or Combi • Gimbal, Head End • Differential Throttling • Moveable Exit Cone	B
11.	Thrust Chamber Cooling	• Regenerative • Film/Barrier • Ablative • Transpiration	B
12.	Combustion Stability Aids	• Delta P • Baffles • Aluminum • Oxidizer Orifice/Granule Size	B
13.	Igniter	• Forward End • Aft End • Pyrotechnic • Hypergolic • Surface	B
14.	Structures	• Isogrid • Semi-Monocoque	C
15.	Avionics, Instrumentation Health Monitoring		C

*Basis: Enable Screening Activities

We Produced Significant Subsystem Screening and Results in 13 of the 15 Subsystem Areas, Including All First and Second Priorities. We Defined the Acceptable Alternatives in the Next Three Sheets. In Only One Case Did We Accidentally Make a Selection by Screening Out All Alternatives Except One. The First Item Below Shows That We Selected Warmed Helium Gas Pressurization for the LO_2 Tank for a Pressure Fed HRB. Conversely, Several Turbopump Fed HRB Tank Pressurization Options Passed the Qualitative Screening. Screening Details, Reasons, and Justifications Are Shown Later in This Section

LEVEL 3 HRB SCREENING RESULTS (PRIORITY A&B)

(Subject Numbers Keyed to Subsystem Options)

1. <u>Tank Pressurization</u>
   TF Only, Cold He, Autogenous (Heated With Dumped Warm Gas) or GO_2 Bleed From Cycle. PF: Warm Gas Dump Heated He

2. <u>Turbine Drive Cycle</u>
   Topping Cycle With Oxidizer Rich Solid-Liquid Preburner or Bleed Cycle With Water Diluted, Stoichiometric Solid Liquid Gas Generator (Cycles 10b or 11a or a Bleed Burnoff Cycle Combination of the Two)

   Drive Cycle/Tank Pressurization Combination: Best Candidate for Conventional Hybrid Seems to Be Topping Cycle 11a With GO_2 Bleed. For SLSC, It Is Dump Heated Autogenous With Bleed Cycle 10b or GO_2 Bleed With Topping Cycle 11a or Bleed Burnoff Cycle

3. & 4. <u>Type and Number of TCAs</u>
   Both Single and Multiple TCAs or Nozzles Will Be Evaluated With One or Four Motor Cases. DeLaval Configurations Will Be Used

---

TF = Turbopump Fed PF = Pressure Fed

LEVEL 3 HRB SCREENING RESULTS (PRIORITY A&B)

Results (Cont.)

5. <u>Solid Propellant Case</u>
Metallic Cases With and Without Composite Hoop Overwrap Will Be Evaluated, With Cylindrical or Conical Shape

6. <u>Solid Grain Design</u>
Use Internal Burning Grain SLSC and Multiperforated or Internal Burning for SSC. Configuration E May Need Multiperforated Design

7. <u>LOx Tank</u>
Composite Overwrapped Metallic Tank With Alternate of Al for Turbopump Fed HRB. Cylindrical or Tapered Shape to Be Evaluated

8. <u>LOx Feed System</u>
SLSC Only - Use Aft Mounted TPA With LO_2 Suction Lines (No Boost Pump). PF Uses All Pressurized Lines. All Lines External to HRB

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LEVEL 3 HRB SCREENING RESULTS (PRIORITY A&B)

Results (Cont.)

9. Thrust Control, MR and PU Control
   Design Solid for Max Q Throttle Down and Up. Use Liquid Control A/R for Thrust Trim Early in Flight and for MR/PU Control Late in HRB Operation. Evaluate Separate MR Control

10. TVC
   Moveable Exit Cone With Option of LITVC

11. Thrust Chamber Cooling
   Ablative or Regenerative With Barrier TCA Cooling as Required, With Ablative, Graphite, or Carbon-Carbon Nozzle Extensions

12. Combustion Stability Aids
   Use All That Are Applicable

13. Igniter
   Forward End Ignition - Pyrotechnic or Hypergolic

Our Screening Was Based Upon the Idea of Meeting NASA Requirements and Priorities With Any Concept We Approved

HYBRID PROPULSION REQUIREMENTS

Our Baseline HRB Was Created to Be STS Compatible

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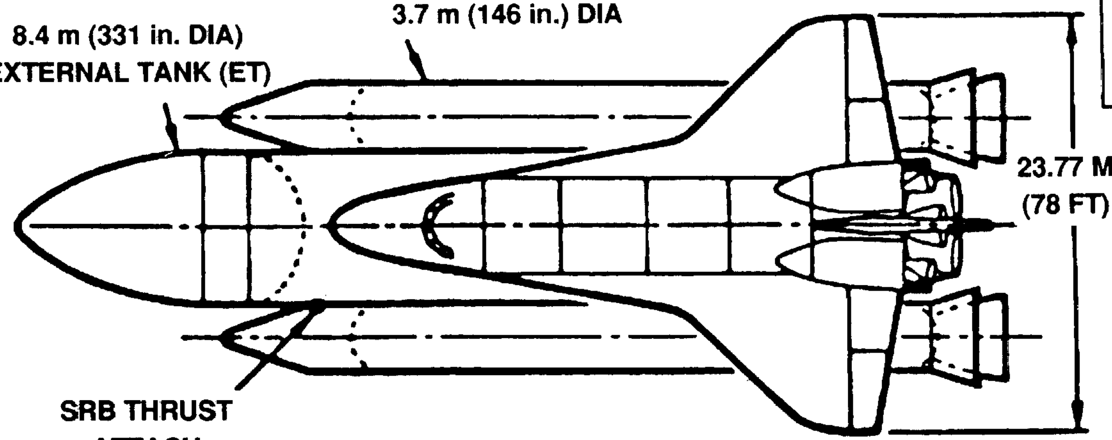
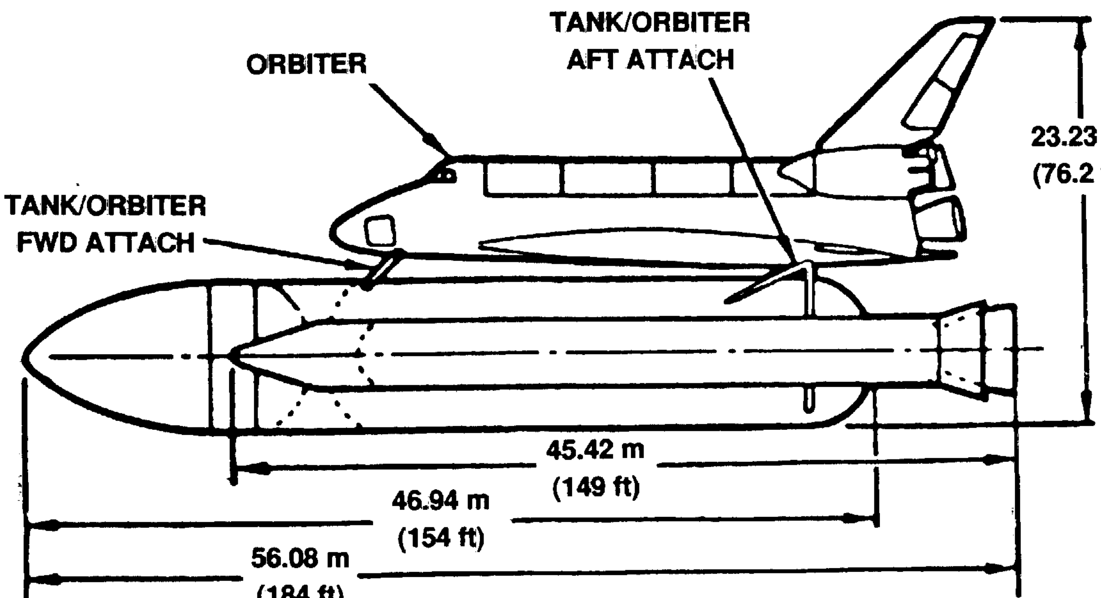
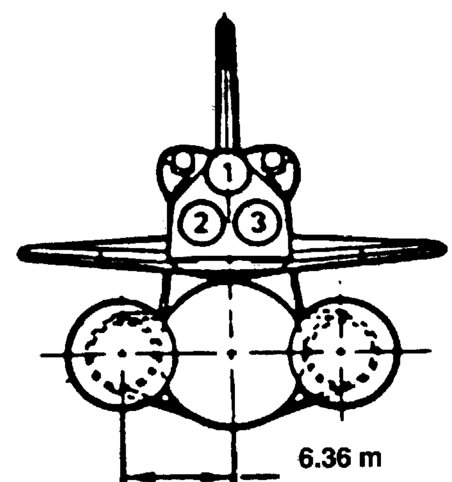
SPACE SHUTTLE VEHICLE



GROSS LIFT-OFF WEIGHT, METRIC TONS 2014 (4,4440K lb) (50 x 100 n MILES BY 28.5 DEGREES)	METRIC TONS (Klb)
SRB	1162 (2562)
ET	737 (1625)
ORBITER	
DRY	69 (152)
CREW AND PROPELLANTS	16.3 (36)
PAYLOAD	29.5 (65)





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The HRBs We Studied All Met STS Booster Performance Requirements

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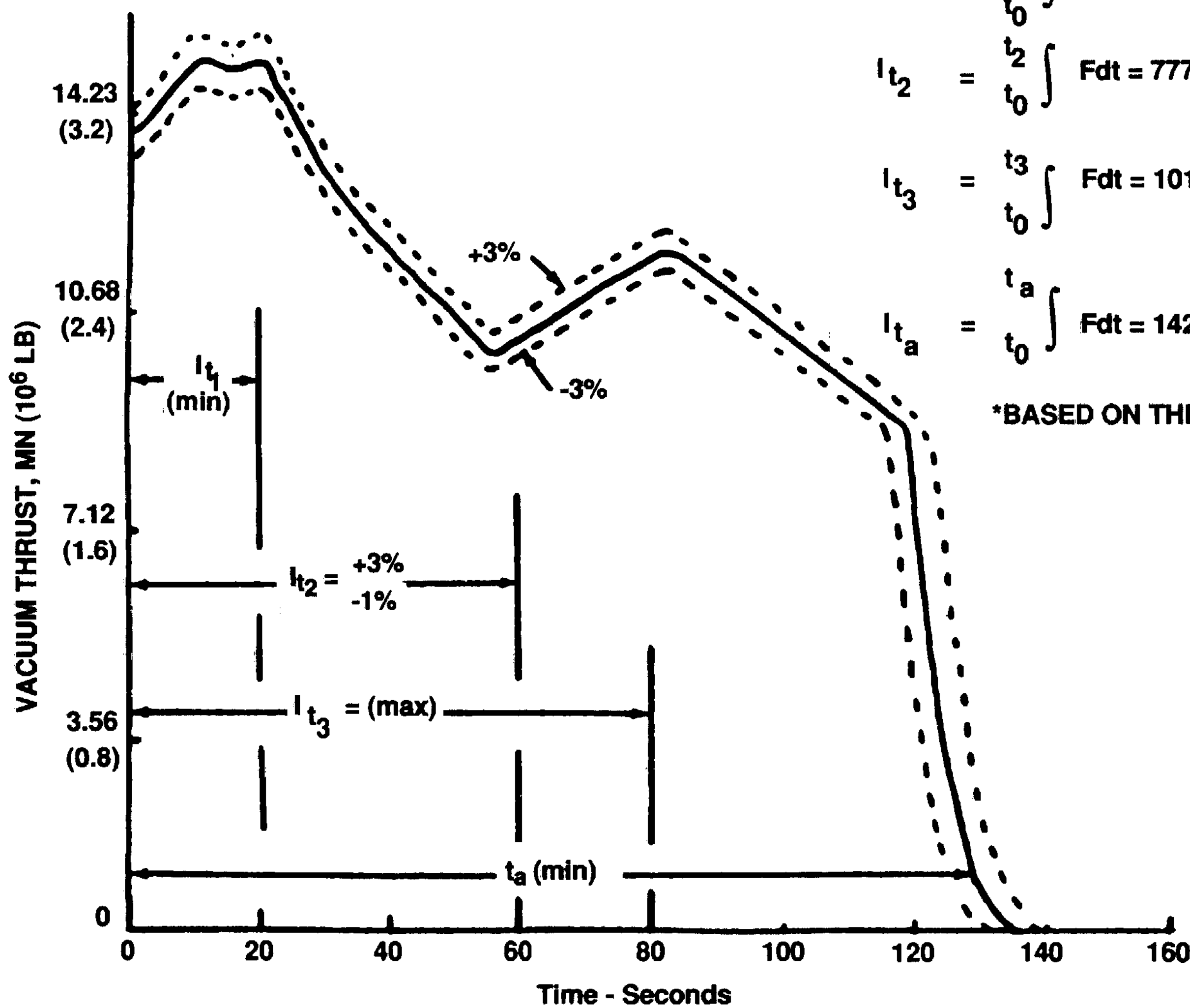
PERFORMANCE SUMMARY



INPULSE VALUES MN-sec (10^6 lb-sec)

$I_{t_1} = \int_{t_0}^{t_1} Fdt = 287.1\ (64.55)\ (MIN)$ $t_1 = 20$ SEC

$I_{t_2} = \int_{t_0}^{t_2} Fdt = 777.8\ (174.86)\ {}^{+3\%}_{-1\%}$ $t_2 = 60$ SEC

$I_{t_3} = \int_{t_0}^{t_3} Fdt = 1017\ (228.66)\ (MAX)$ $t_3 = 80$ sec

$I_{t_a} = \int_{t_0}^{t_a} Fdt = 1424\ (320.15)\ (MIN)$ t_a = ACTION TIME

T_0 = IGNITION

*BASED ON THRUST-TIME TRACE IN TABLE 1

ASRM REFERENCE VACUUM THRUST-TIME TRACE AND IMPULSE VALUES AT 60°F

Our Approach Included an Early Yes/No Type Qualitative Screening of Developed Concepts and a Subsequent Quantitative Selection, Based on Scores Computed With Life Cycle Cost and Payload to LEO Data. Some of Your Priorities Were Considered in the Screening Process and Some During Selection. All of Your Requirements Were Used to Screen Out Unworkable Concepts Early on

This Chart Explains How the Task 1 Screening Criteria Contained All of NASA's Requirements and Priorities, Except Life Cycle Cost and Payload Performance, Which Are Included in Subsequent Selection Activities in Task 3. In Addition, We Added the Umbrella Category, Availability, to Our Criteria Under NASA's Fourth Priority of "Other"

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AEROJET STUDY CRITERIA CONCUR WITH NASA MSFC HRB PRIORITIES

NASA Priorities for HRB (ranked)

1. Flight **Safety** and **Reliability**
2. Life Cycle Cost
3. Performance **(PAYLOAD)**
4. Other

Task 1

Qualitative (yes/no) Screening (not ranked)

I. **Safety**
   a. Explosive Hazard
   b. Auto-Ignition Hazard
   c. Toxicity
   d. FMEA - Loss of STS
   e. Physical

II. Availability
   a. Tech Demo Date
   b. Development Risk
   c. Producability
   d. Maintainability
   e. Reliability - Loss of Mission

III. Design and Operational STS Compatibility **Requirements**
   a. Geometric
   b. Operational

Task 3

Quantitative Selection

Loss of Mission **Reliability** = **LCC** Input

↓

LCC = Selection Fig. **Of Merit**

↑

STS **Payload** = a **FOM** element

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HYBRID PROPULSION QUALITATIVE CRITERIA

- Safety
  - FMEA
- Availability
- Design and Operation
  - Requirements

We Identified Five Subcategories Under the Safety Heading in the First Sheet of Our Three-Page Screening Criteria. Our Format Included Scope, Reason for the Scope Items, the Qualitative Value Against Which We Screened, and the Basis for the Value, Where Not Completely Obvious. Subcategory D, Catastrophic Failure Modes, Is Amplified on Pages 19 and 20, Following

HYBRID ROCKET BOOSTER (HRB)

HRB FOR STS STUDY (Yes/No) SCREENING CRITERIA

All Candidates Must Meet STS (ASRM) Performance Requirements

SCOPE	REASON	SCREENING VALUE	BASIS
I. SAFETY (propellant & ignition)	Preserve lives, facilities, & vehicles	See below	See below
A. Explosive hazard	"	Class 1.3 or 1.4	Typical of man rated system
B. Auto ignition hazard			
1. Shock/vibration sensitivity		Impact energy	Need to control use
2. Thermal sensitivity		Autoignition temperature	Need to control use
3. Radiation sensitivity		________	
4. Hypergolicity with LO2		No	Autoignition
C. Toxicity 1. Crew		No poisonous vapors	Maximize Public and crew safety & minimize Environmental Impact
2. Environmental a. Exhaust toxicity		No Cl, F, Be NOx, or O3	Proven toxins
b. Exhaust corrosivity		No Cl, F, Be	Proven corrosives
c. Chemical attack/ contamination		Allow safe operation	
d. Biosphere pollution by gas, liquid, or solid waste. No asbestos.		None	
D. Catastrophic Failure Modes		Minimize modes	Maximize Safety
1. Case field joints		None	Preclude hot, high pressure gas leaks
2. Clogging of downstream components		No	Prevent case over-pressure
3. Fratricidal protection		Yes	Prevent secondary failures
4. Adequate grain burn area control		Yes	Prevent case over-pressure
E. Physical			
1. Leaks (HP or chemical)		No	
2. HP ruptures		No	
3. Thermal distortions		No	
4. Natural environment induced hazards		No	

The Category "Availability" Included All Five Fundamental Reasons Why the Payload Could Not Be Available in LEO When Required: The Technology Couldn't Be Demonstrated for the HRB on Schedule; the HRB Couldn't Be Developed in Time; It Couldn't Be Produced Fast Enough; It Was Down for Extended Maintenance When Needed for Flight; or Couldn't Be Relied Upon to Deliver Payload

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HYBRID ROCKET BOOSTER (HRB)

HRB FOR STS STUDY (Yes/No) SCREENING CRITERIA

All Candidates Must Meet STS Performance Requirements

SCOPE	REASON	SCREENING VALUE	BASIS
II. AVAILABILITY	Fundamental need for operational program	See below	See below
A. Technology Aquis'n Date	Technology must be demonstrated	Sept 1995	Post ASRM use
B. Development Risk	Program risk must be acceptable	Medium	Low = fully devel'd Med = tech demons'd High = scientifically feasible
C. Producibility	Processes & mat'l must be usable	Proc.&mat'l available at Sept 1999	Fundamental need (IOC = Sept 2002)
D. Maintain'ity	HRB must be useable on demand	Accessible LRU's and condition monitor	Must establish maintenance need and enable it with minimum delay & cost concept
1. Maximize shelf life		Yes	
2. Accessible & visible hardware		Yes	Correlates with maintenance cost
3. Damage during installation		No	" " " "
4. Sequential assembly required		No	" " " "
5. Unrealistic facility requirements or maintenance envi'nt		No	" " " "
6. Special handling required		No	" " " "
7. Periodic maintenance required		Yes	" " " "
E. Reliability	Successful mission need	-----	
1. Design complexity		Minimum	Correllates with reliability
2. Experience in field		Required outside of new technology areas	" " " "
3. Redundancy for loss of mission failure modes		Yes	Prevent loss of mission

The Design and Operation Criteria Assured That the STS Requirements, Both Operational and Interface, Could Be Met by All Successful Candidates

HYBRID ROCKET BOOSTER (HRB)

HRB For STS Study (Yes/No) Screening Criteria All Candidates Must Meet STS Performance Requirements

Scope	Reason	Screening Value	Bases
III. Design & Operation	Complete Compatibility With STS	See Below	See Below
A. Geometric			
1. Diameter, Max., m (In.)		3.81 (150.) 4.95(195.)	ASRM LRB Study
2. Length, Max., m (In.)		38.48 (1,514) 47.00 (1,850)	ASRM LRB
3. Thrust Takeout Location (FTL), m (Ft.) Above MLP		38.07 (124.9)	STS Design
4. Stiffness Below FTL		Near SRB	ET Design Launch and Flight Loads
5. Nozzle Exit Dia.		41.37 kPa (6. psia) Size	New MLP Potential (LRB Study)
B. Operational			
1. Feasible Start Sequence		Yes	
2. Throttleable		1.5 to 1	
3. TVC		+/- 6 deg	SRM, ASRM
4. Ignition, Time to 90% PC		TBD sec	Operation Requirement
5. Solid Burning Rate, cm/sec (ips.)		1.0 to 1.5 (0.4 to 0.6)	
6. Extinguishable		Yes	Abort Requirement
7. Vibration Control		Yes	STS Interface
8. Thermal Control		Yes	STS Interface
9. Thrust, Total Impulse, Burn Time Requirements Met		Yes	ASRM Requirements

These Two Sheets Augment the FMEA Considerations of Criteria I.D. Eleven Loss of STS-Inducing Failures Were Examined and Optional Solutions Identified. These Were Used in Yes/No Screening for Criterion I.D.

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HRB STUDY—CRITICAL FAILURES CAT. 1 = LOSS OF VEHICLE

No.	Failure	Result	Solutions
1.	Loss of TPA Buffer Seal Flow	Explosion in TPA Due to Fuel and Oxidizer Mixing	a. Use No Fuel - Oxidizer Seals b. Assure No Loss of Flow
2.	Low Temperature GO_2 Turbine	O_2/Metal Ignition in TPA Due to High Temperature Generation by Rub of Bearing, Seal, or Tip	a. No O_2 Turbines b. Tested, Fully Compatible Designs and Materials (Monel, Silver, Hybrid Bearings)
3.	Motor Case Exhaust Plugging	Motor Case Overpressure and Rupture	a. Throat Only Downstream b. Radial Injector and Throat Downstream c. No Large Solids, Sticky Fine Particles or Condensible Liquids Generated by Grain or Igniter d. No Multiple TCA Warm Gas Shutoff
4.	Adequate Grain Burning Area Control	Motor Case Overpressure and Rupture	a. Good Propellant Physical Properties b. Non-Self Sustaining Solid Propellant (Class 1.4)
5.	Leakage of Motor Case	Explosion of ET Propellant	a. No Motor Case Joints b. New (Bolted) Joint Concept
6.	Lack of Fratricidal Protection	Domino Loss of All Propulsion or Explosion	a. Shielding b. Proper Locations

Note: GG Loss of Diluent or Off-MR Operation Is Category 2 Failure (Engine Out)

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HRB STUDY—CRITICAL FAILURES CAT. 1 = LOSS OF VEHICLE (Cont)

No.	Failure	Result	Solutions
7.	Pressurant Contaminant in LO_2 Tank	Explosion of LO_2 Tank	Pure Inert Gas or GO_2
8.	Hot Pressurant in LO_2 Tank	Rupture of LO_2 Tank	Limit Pressurant Inlet Temperature
9.	Large, High Pressure Gas Bottle Break	Explosive and Fragment Damage to Vehicle, Payload, Facility, Personnel; Secondary Explosions	No Large, High Pressure Gas Bottles
10.	Combustion Instability	Explosive TCA Loss	a. Demonstrated High-Frequency and Chug Stability b. Fratricide Protection for Vehicle
11.	Loss of HRB TVC	Loss of Mission or Loss of Vehicle	Reliable TVC

In the Following Section We Present the Methodology Used and Results Obtained for Generating Hybrid Concepts Along With the Screening for Each as a Practical Way of Achieving Clarity in Our Data Presentation. We Began Our Study by Considering Solid, Hybrid, and Liquid Propellant Processes, Because as We Will Show, Efficient, Repeatable Combustion Is an Important Key to Safe, Low Cost, High Performance Boosters

HYBRID PROPULSION CANDIDATE CONCEPT

- Generation
- Screening

Solid Propellant Motors Burn Premixed Propellants. Combustion Heat Vaporizes Fuel and Oxidizer From the Grain Surface Together and Ignites Them So They Burn Efficiently and Quite Completely.

Problems With Solid Boosters Lie in Areas Other Than Combustion as Noted at the Bottom of the Chart

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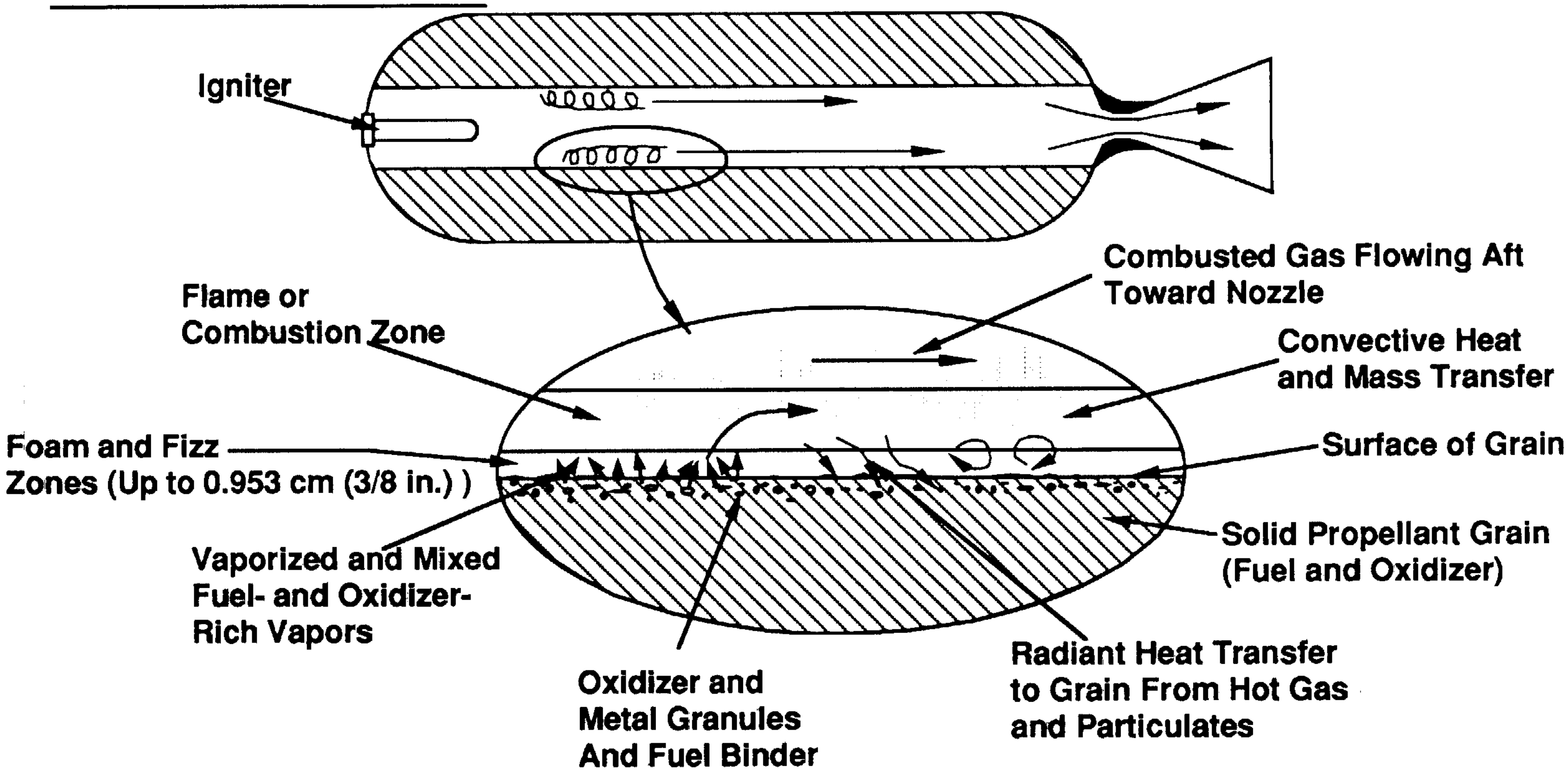
HOW DO THE COMBUSTORS WORK AND WHAT ARE THE PROBLEMS ?

STEP A : Solid Combustion



- Premixed Fuel and Oxidizer Vapors Rise From the Grain Surface Due to Heat Transfer From the Adjacent Combustion Zone

- Problems,
  - Lack of Control
  - Premixed Propellants Cast in Place
  - Low Specific Impulse Due to Use of Solid Ozxidizers

3.16.0.7

With Conventional Hybrid Motors, Oxidizer Is Injected at the Front of the Fuel Grain, and Combustion Heat Drives Fuel Vapors From the Surface of the Grain. Combustion Heat Also Vaporizes the Oxidizer in the Center of the Grain Port(s). Complete and Efficient Combustion Can Be Obtained Only by Complete Mixing of These Two Reactants in the Port Volume. This Is Unlikely in a Large Motor, Because of the Growing Thickness of the Interstitial Combustion Gas Layer Flowing Toward the Nozzle

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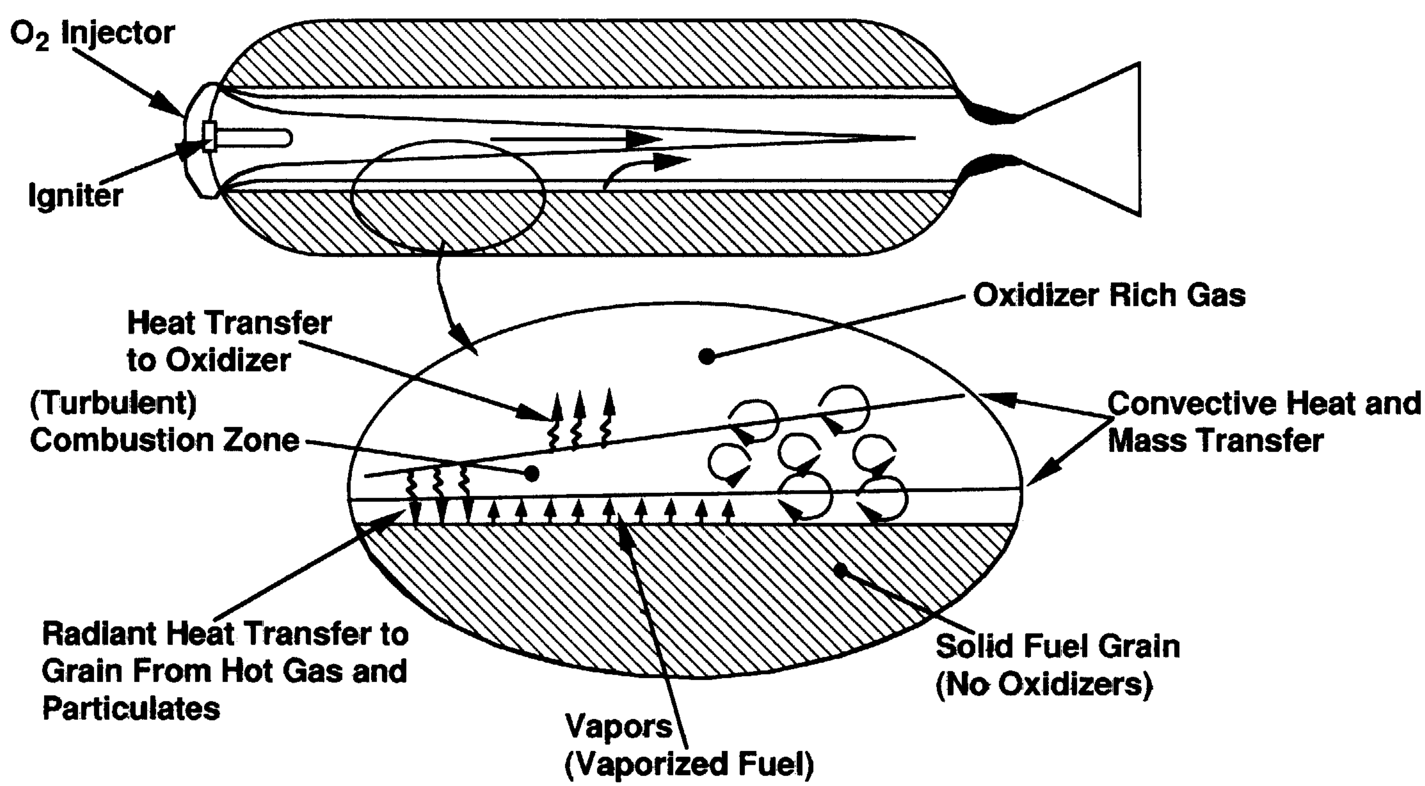
HOW DO THE COMBUSTORS WORK AND WHAT ARE THE PROBLEMS ?

Hybrid Combustion



- Fuel Vapors Rise From the Grain Surface
- Oxidizer Vapors Flow Down the Center of the Port
- Turbulent Mixing Combines Them in the Combustion Zone

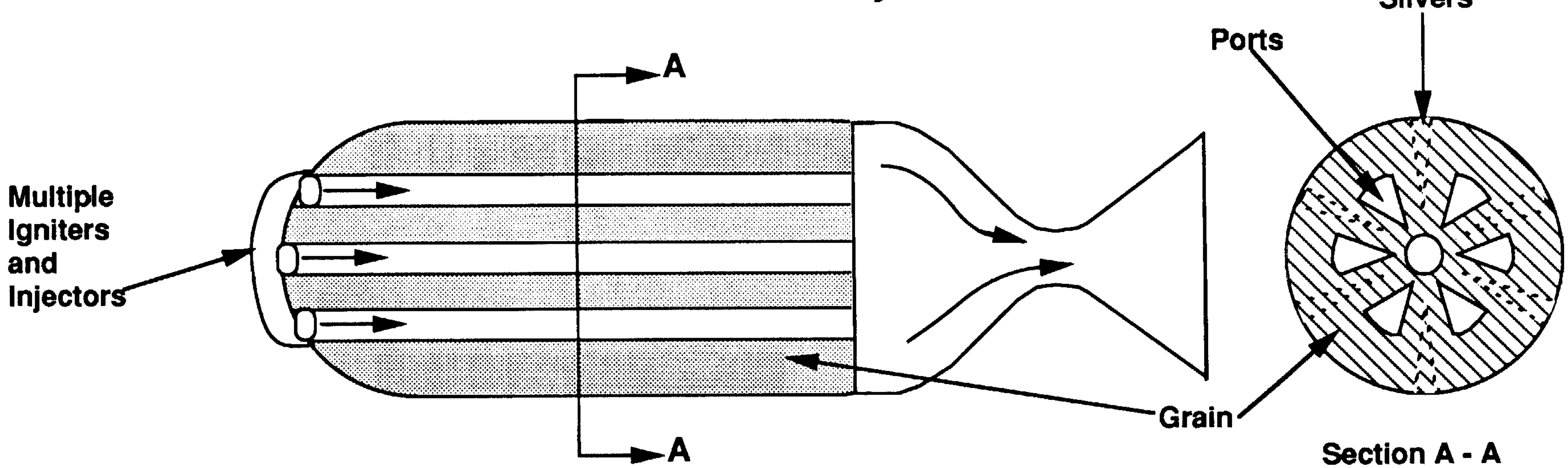
Problem: Large Ports Are Mixing Limited and Have Poor Combustion Efficiency

Multiport Grains Help Solve This Problem by Making a Large Hybrid Look Like a Cluster of Smaller Ones, but:

- It Creates Solid Propellant Sliver Losses
- It Complicates Injectors, and
- It Does Not Solve the Mixing Problem Completely

POTENTIAL SOLUTIONS

Multi-Port Hybrid



- Multiport Grains Allow Larger Motors to Operate With the Combustion Efficiency Approaching Smaller Units
- Sliver Losses, However, Typically 5 % or Higher Result In a Large Motor, This Is a Large Weight Penalty, and Slivers Break up Near Burnout

3.16.0.3

Another Combustion Efficiency Aid Is to Add Some Solid Oxidizer to the Fuel Grain. This Also Has Some Disadvantages and Does Not Solve the Mixing Problem Completely

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POTENTIAL SOLUTIONS QUASI-HYBRID

(Add Some Solid Oxidizer to the Fuel Grain)

- This Concept May Allow Larger-Ported Motors to Attain Combustion Efficiencies Near Those of Smaller Units by Adding Turbulence to the Mixing Process via Combustion Near the Surface of the Grain, and by Reducing the Total Amount of Port Mixing Required

- Use of Excess Solid Oxidizer Reduces Theoretical Isp and the Liquid/Solid Weight Ratio

- Much Port Mixing Still Required for Complete Combustion

Combinations of the Two Introduce Many Issues and Do Not Solve the Mixing Problem Effectively or Completely for Large Hybrids.

We Decided to Look at Liquid Propulsion Combustion Technology to See if a More Effective Solution Could Be Found

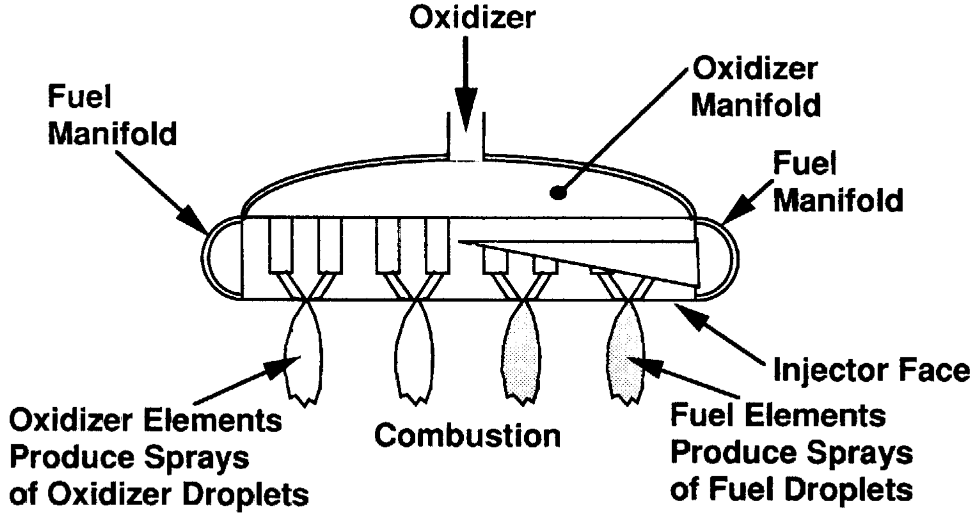
LARGE HYBRIDS - MULTIPORT QUASI-HYBRID

- Very Large Hybrid Motors May Be Made to Work Using Only a Few Ports Within a Quasi-Hybrid Grain. Mixing, Slivering and Solid Oxidizer Induced Theoretical I_{sp} Losses all Result
- Alternatively, Liquid Rocket Injection Technology Could Be Applied to Solve the Hybrid Mixing Problem

Effective Combustion of Liquid Propellants Is Created by the Impingement of the Liquid Streams (Oxidizer and Fuel). The Resultant Spray Contains Fine Droplets That Mingle as They Are Heated and Vaporized by the Nearby Combustion Heat. Combustion Is Typically Uniform, Complete, and Efficient, Because the Local Conditions Are Controlled and Uniform Across the Injector Face

Combustion Instability Limits Our Ability to Achieve Very High Efficiency in Very Large Combustors, Even With Good Damping Devices

LIQUID COMBUSTION



1. Liquid-Liquid (Typically, Multi-Orificed Injector)
   - Droplets Vaporize From Combustor Heat,
   - Vapors Mix Due to Injection Pattern (Premix Due to Injector Design) and Combustion Turbulence, and Combust Rather Uniformly.

Problem: Combustion Instability Limits the Orifice Sizes and Combustion Efficiency Attainable for Any Large System

Combusting a Gas With a Liquid Reduces the Severity of the Instability Problem, Allowing Higher Performance in Large Combustors. Gas/Liquid Injectors Form Liquid Droplets by Impinging Liquid Streams Like Liquid Injectors Do, So They Mingle With the Other, Gaseous Reactant Being Injected. Alternatively, They Can Depend Upon Large Differences Between Gas and Liquid Injectant Velocities to Cause Turbulent Shear Mixing at Each Gas/Liquid Interface, or Both Methods Can Be Used Together

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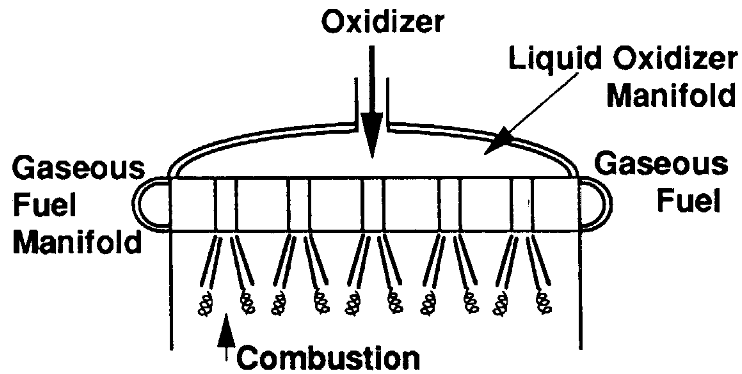
GAS - LIQUID COMBUSTION



Many Bipropellant Sprays

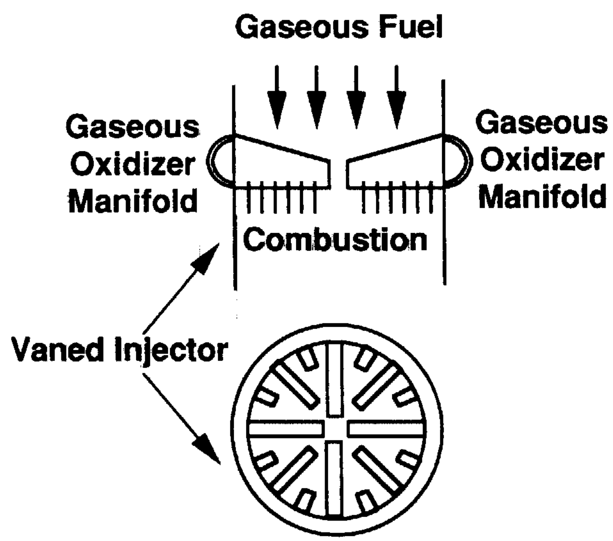
(Typically Co-axial Injector)

- Fuel Heat and Combustion Heat Vaporize the Oxidizer
- Fuel and Oxidizer Vapors Mix Due to Premix Effect of Injector Design, Differential Injection Velocities and Combustion Turbulence
- Combustion Instability Problems Are Reduced (vs. Liquid-Liquid) Because of Fewer (and Reduced Severity) of Combustion Rate Affecting Feedback Mechanisms

Two Gases Can Be Reacted Also, and the Greatest Combustion Stability Is Expected From This Concept. However, It Is Difficult to Cause One Gas Stream to Penetrate Another, and Mixing Is Usually Thought to Be the Limitation With Gas/Gas Injectors. This Can Lead to Lower Combustion Efficiency, Regardless of the Injector Type, Although Some Are Better Than Others for Large Combustors

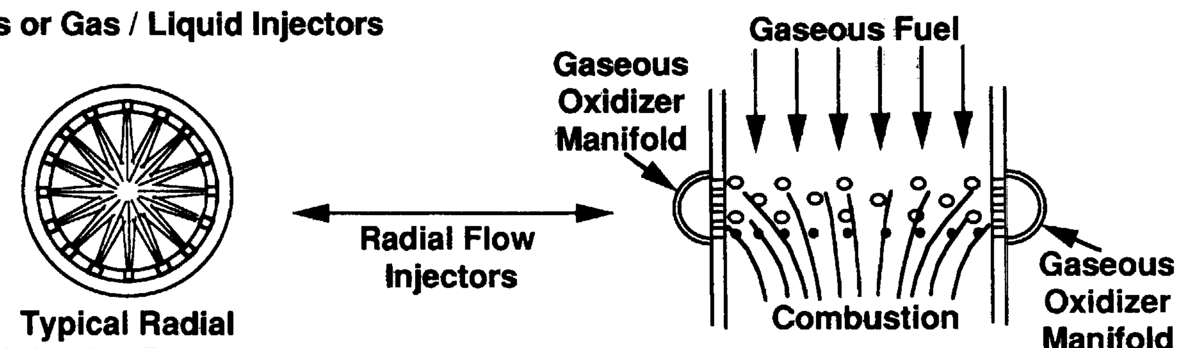
POTENTIAL SOLUTIONS

Gas - Gas Combustion
(Typically Vaned Or Radial Flow Injector)



Typical Vane Pattern
for Gas-Gas or Gas / Liquid Injectors

- No Vaporization Process Is Required
- Mixing Occurs as a Result of Injector Design Premix Effect and Combustion Turbulence, and It Is Limited by Gas Jet Penetration Problems
- Minimum Combustion Stability Problems Exist Due to Minimum Number of and Least Severity of Combustion Rate Feedback Mechanism



Typical Radial
Injection Pattern

Feature: No Hardware in Fuel Gas Stream
Problem: Mixing Efficiency

In the Aerojet Solid/Liquid Staged Combustion Concept, Gas/Liquid (or Gas/Gas) Injector(s) Are Used to Assure Stable High Combustion Efficiency and HRB Performance. In This Concept the Solid Case Is a Fuel Rich Gas Generator for the Aft-Mounted Gas/(Liquid) Oxidizer Injector. The Fuel-Rich Solid Propellant Is Ignitable, Self Sustaining (Under High Pressure) and Extinguishes at Low Pressure. The LO_2 and Fuel-Rich Warm Gases Are Burned Off in a Liquid Rocket-Type Thrust Chamber, Which May Use Solid Propulsion Technology in the Nozzle

Subsequent Studies Showed That the Benefits of Efficient Combustion Outweigh the Theoretical Loss in Performance Caused by Using Some Solid Oxidizer in the Solid Grain

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POTENTIAL SOLUTIONS

Solid - Liquid Staged Combustion

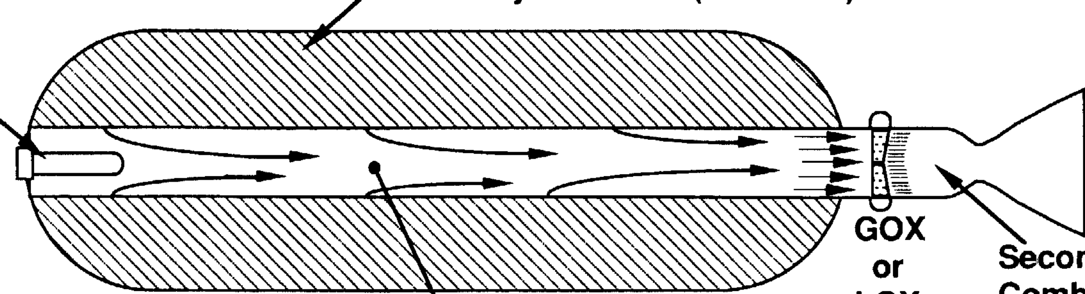
An Alternative Approach to The Use of Multiport Grain Design (with Residual Sliver Losses) and The Quasi - Hybrid Grain Is:

Quasi - Hybrid Solid Gas Generation and Downstream (Aft) Oxidizer Injection. Combustion Efficiency Is Controlled by the Mixing Provided by the Gas - Liquid or Gas - Gas Injector

Cryogenic Vaned Injectors Need to Be Compatible With Primary Combustor Grain Design and Propellant Gases

Radial Flow Injectors Limit the Gaseous Fuel Velocity to Provide Oxidizer Stream Penetration in Large Diameter Injectors and Mixing Efficiency

Either Design Is Feasible, But the Vaned Injector has Much Better Performance Potential



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Our Level 1 and 2 Screening Tasks Included:

- Inject Fuel or Oxidizer
- Combustion Scheme - Method and Location of Injection
- Solid Propellant Type

For Continuity and Clarity Reasons, the Reader May Wish to Skip to Pages 44-47 for Level 2 Screening Information First (Combustion Scheme) and Return to Pages 32-43 for Level 1 Propellant-Type Screening, Before Continuing With Level 3 (Subsystems) Screening

LEVEL 1 AND 2 SCREENING TASKS

No.	Subject	Option	Priority* Assigned
1.	Propellant Injected	• Oxidizer • Fuel	A
2.	Liquid Propellant Injection Method	• Forward End • Aft End • Forward and Aft • Liquid or Gas	A (A = Mandatory, B = Secondary Importance C = As Required)
3.	Solid Propellant	• Pure Fuel or Oxidizer • Very Fuel or Oxidizer Rich	A

*Basis: Enable Screening Activities
HRB Study Screening Levels:

Level 1 - Hybrid Propellant Physical and Chemical Types
Level 2 - Combustion Methods
Level 3 - Subsystems for HRB Program

LEVEL 1
PROPELLANTS

This Concept Generation Matrix Identified 40 Propellant Combination Candidates by Matrixing Solid Propellant Types Versus Injectable Propellant Types. The Lower Left and Upper Right Quadrants Do Not React and Were Rejected. The Lower Right Quadrant Probably Will Not Meet Reasonable Size and Weight Limits and Will Not Extinguish. We Know of No Oxidizer-Only Grain Compositions, and Oxidizer-Rich or Maximum Performance Grains That Will Extinguish Easily or at All. The Eight Candidates at the Upper Left Corner Appear Feasible for HRB Application

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LEVEL 1—PROPELLANT TYPES CONCEPT GENERATION MATRIX

Solid Composition

Injectant	Fuel Only	Fuel Rich	Maximum Performance	Oxidizer Rich	Oxidizer Only
Neat Liquid Oxidizer	1	2	No Reaction	No Reaction	No Reaction
Gelled Oxidizer	3	4	No Reaction	No Reaction	No Reaction
Gelled Oxidizer With Additives	5	6	No Reaction	No Reaction	No Reaction
Gaseous Oxidizer	7	8	No Reaction	No Reaction	No Reaction
Gaseous Fuel	No Reaction	No Reaction	9 Reduce MW, Large Size, or Does Not Distinguish	10 Low ρB = Large Size	11 Low ρB = Large Size
Neat Fuel	No Reaction	No Reaction	No Reaction	12 Low I_{sp} + Low ρ_B = Large Size	13 Low I_{sp} + Low ρ_B = Large Size
Gelled Fuel	No Reaction	No Reaction	No Reaction	14 Low I_{sp} + Low ρ_B = Large Size	15 Low I_{sp} + Low ρ_B = Large Size
Gelled Fuel With Additives	No Reaction	No Reaction	No Reaction	16 Low I_{sp} + Low ρ_B = Large Size	17 Low I_{sp} + Low ρ_B = Large Size

The Formal Screening Process Involved Use of a Chart Summarizing Our Screening Criteria Categories on the Left Versus Candidates Listed Across the Chart. In This Case, Fuel Injection Was Screened Out Because Oxidizer-Rich Grain Hybrids Will Not Extinguish Well or at All, an Operational Requirement. A Single "No" Screens Out Any Concept

Therefore, We Screened to Liquid Oxidizers and Solid Fuels

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WE SCREENED OUT FUEL INJECTION

Level 1	Injectant	
Screening Criteria	**Oxidizer**	**Fuel**
I. Safety		
A. Explosive Hazard		
B. Autoignition		
C. Toxicity		
D. FMEA		
E. Physical		
II. Availability		
A. Technology Acquisition		
B. Development Risk		
C. Producibility		
D. Maintainability		
E. Reliability		
III. Design and Operation		
A. Geometric		Low I_{sp}
B. Operational		Not Extinguished
Result	Yes	No

The LO_2 Additives That Were Considered Would Gel the Oxidizer and Increase Its Density. Liquids Sprayed Into the LO_2 Would Freeze Into Small Particles, Gel the Mixture, and Hold Themselves in Suspension

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LIQUID OXIDIZER CANDIDATES WERE CONSIDERED

Neat Liquid Oxidizers	Oxidizers Gelled With Additives
Oxygen Compounds	
O_2	$O_2 + H_2O_2(S)$
H_2O_2	
N_2O_4	$O_2 + N_2O_4(S)$
O_3	
Halogenated Compounds	
F_2	
ClF_5	
ClF_3	
NF_3	
N_2F_4	
Halogen and Oxygen	
FLOX	
OF_2	
ClO_3F	

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LIQUID OXIDIZER SCREENING RESULTS

Screened Out	Basis
All Gaseous	Geometric and Performance
All Gelled Without Additives	Technology Acquisition—No Benefit
All Halogenated	Environmental, Toxicity
N_2O_4	Toxicity
O_3	Safety—Explosive, Unstable
H_2O_2	Safety—Occasionally Explosive When Pressurized

Remaining Candidates	
O_2	
$O_2 + H_2O_2(S)$	(Unknown Safety Characteristics)
$O_2 + N_2O_4(S)$	

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WE SCREENED OUT H_2O_2 FOR HRB BECAUSE:

Although Its PROS Are:

- High Performance (I_{sp} and Density)
- Compatible With Clean, Vented Systems of Proper Materials

The CONS of Experience Are:

- Explosive Decomposition in Pressurized Systems
- No Large Operational System With High Purity H_2O_2
- Armed Forces Eliminated H_2O_2 From All Systems
- Facility Issues at JFK-FC

PROPERTIES OF CANDIDATE OXIDIZERS

	LO_2	$LO_2 + H_2O_2(S)$*	$LO_2 + N_2O_4(S)$*
Normal Boiling Point, °K (°F)	91 (-297)	91 (-297)	91 (-297)
Density, g/cc at Temperature, °K (°F)	1.14 91 (-297)	1.47 91 (-297)	1.55 91 (-297)
Heat of Formation, (cal/gm liq. at 298°K)	-96	-940	-124
Freezing Temperature, °K (°F)	54 (-362)	54 (-362)	54 (-362)

*50-50 Volume Percent Mixture

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OBJECTIVES

- Define Fuel Rich and Fuel Only Solid Propellants Suitable for Hybrid Rocket Booster Application:

Characteristics	Solid/Liquid Staged Combustion	Head End LOx Injection
Tc at 6895 kPa (1,000 psi)	<1111°K (1,540°F) Without LO_2	Maximum
HCl Exhaust	< 1% Total	< 1% Total
Burning Rate at 6895 kPa (1,000 psi)	>0.9 cm/sec (0.35 in./sec)	>0.9 cm/sec (0.35 in./sec)
Density	Maximum	Maximum
Hazard	Class 1.3, 1.4	Class 1.3, 1.4
Extinguishment Capability	Yes	Yes
Performance	Maximum	Maximum
Exhaust Condensables Without LO_2	Minimum	--

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SOLID PROPELLANTS CONSIDERED

- Fuel Rich
  - Baseline ANB-3302-4
    - AP/AS/HTPB
  - Nonmetallized
    - AN/HTPB/PAMS
  - Metallized
    - AN/Al/HTPB
    - AN/B/HTPB
  - Scavenger
    - AP/$NaNO_3$/HTPB
  - Bipropellant Grain
- Fuel Only
  - Similar to Fuel Rich Without Oxidizer or Scavenger

AN or Scavenger Required to Meet Low HCl Goal

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PROPELLANT CANDIDATES

No.	Solid Propellant	Oxidizer	Grain Density, kg/m^3 (lbm/ft^3)		Isv 1800/6, sec	Solid Propellant Ingredients Cost, $/kg ($/lbm)	
3	PAMS/BAMO	LO_2	1214	(75.8)	306.2	$6.02	($2.73)
6	HTPB HC/Al	LO_2	1627	(101.6)	280.4	4.30	(1.95)
7	HTPB HC/B	LO_2	1517	(94.7)	307.1	57.61	(26.13)
8	PEBC	LO_2	1097	(68.5)	311.2	1.79	(0.81)
9	BAMO/Al	LO_2	1957	(122.2)	260.1	19.82	(8.99)
10	VOX/BAMO/Al/ PEBC	LO_2	1664	(103.9)	277.5	8.27	(3.75)
11	PEBC/Al/BAMO	$LO_2 + N_2O_4(S)$	1664	(103.9)	272.9	8.27	(3.75)
12	PEBC/30% B	$LO_2 + N_2O_4(S)$	1518	(94.8)	297.2	57.61	(26.13)
13	PEBC/30% B	$LO_2 + H_2O_2(S)$	1518	(94.8)	304.6	57.61	(26.13)
14	PEBC/Al/BAMO	$LO_2 + H_2O_2(S)$	1664	(103.9)	283.6	8.27	(3.75)
Fuel Only							
8B	PEBC	LO_2	876	(54.7)	320.3	2.09	(0.95)
8C	PEBC/35% Al	LO_2	1147	(71.6)	308.9	4.34	(1.97)
Baseline	ANB3302-4	LO_2	1539	(96.1)	273.0	2.93	(1.33)

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PROPELLANT COMPOSITIONS— FUEL RICH

Propellant Ingredients Wt, %	Reference Propellant (ASRAM)	Hybrid Propellants				
				Metallized		
		Baseline ANB-3302-4	Non-Metallized PAMS	Al	B	Scavenger
Ammonium Perchlorate Oxidizer (AP)	69.0	61.5	3.2	3.2	3.2	20.31
Ammonium Nitrate Oxidizer (AN)	--	--	39.8	39.8	39.8	--
Sodium Nitrate Oxidizer ($NaNO_3$)	--	--	--	--	--	14.69
Ammonium Sulfate Coolant (AS)	--	16.0	--	--	--	--
Poly and Methyl-styrene Fuel (PAMS)	--	--	35.0	--	5.0	
Aluminum Fuel (Al)	19.0	--	--	35	--	
Boron Fuel (B)	--	--	--	--	30	
HTPB Binder	12.0	22.5	22.0	22.0	22.0	65.00
Thermoplas-tic Elastomer (TPE)	--	--	--	--	--	
	100.0	100.0	100.0	100.0	100.0	100.0

Fuel Rich Compositions Identified for Hybrid Application

PROPELLANT PROPERTIES— FUEL RICH

		Hybrid Propellants				
				Metallized		
Propellant Property	Reference Propellant (ASRAM)	Baseline ANB-3302-4	Non-Metallized PAMS	Al	B	Scavenger
Tc at 6895 kPa (1000 psia), °K (°F)	3567 (5,961)	1453 (2,155)	1178 (1,660)	2629 (4,272)	2232 (3,558)	1074 (1,473)
Density, g/cc	1.804	1.540	1.229	1.628	1.519	1.098
I_{sps} (LOX), sec	263.6	262.5	287.3	278.5	283.3	290.4
Density (LOX), g/cc	1.804	1.384	1.177	1.448	1.302	1.124
HCl, %	20.9	13.0	0.31	~0.31	~0.31	0.78*
Hazard Class	1.3	1.4	1.4	1.4	1.4	1.4
Exhaust Condensed Phase	Al_2O_3	No	Carbon**	Al_2O_3	B_2O_3	NaCl

*Neutralized by Na, NaOH
**May Get Gaseous Products

Hybrid Propellant I_{sv} Is Greater Than Current Shuttle, but Density Is Lower. Metallized Compositions Significantly Raise Tc and Density. Low HCl Achieved With AN Oxidizer or Scavenger Concept.

LEVEL 2
COMBUSTION

This Concept Generation Matrix Identified Six Combustion Scheme Concepts by Matrixing Three Types of Solid Propellant Versus Two Combustion Schemes, Single and Two-Staged Combustion. Although There Is an Empty Location in the Matrix—You Cannot Reburn Maximum Performance Solid Exhaust to Any Significant Advantage—There Are Two Possibilities for Fuel-Rich Grain Recombustion. The Solid Case Can Be a Stand-Alone Fuel-Rich Solid Propellant Gas Generator (Case D) or It Could Have LO_2 Head-End Injection and Be a Hybrid Gas Generator. In the Latter Case (E) the HRB Would Have Two LO_2 Injectors, One at Each End of the Solid Case

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COMBUSTION METHODS CHART

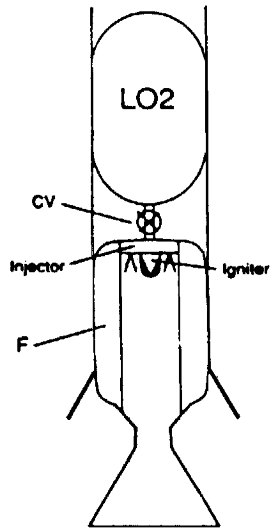
	Two Stage Combustion	Single Stage Combustion
I Fuel Only Solid Grain	C Conventional Hybrid GG With Gas/Liquid Secondary TCA	A Conventional Hybrid
II Fuel Rich Solid Grain	D Fuel Rich Solid GG With Gas/Liquid Secondary TCA (SLSC) E Quasi-Hybrid GG With Gas/Liquid Secondary TCA	B Conventional Hybrid With Enhanced ERE (Quasi-Hybrid)
III Max I_{sp} Grain	Nil Reaction; Hot Injector and Complex	F Direct Injection of Hydrogen or Other Low Molecular Weight Gas

NOTE: Oxidizer Injection May Be Liquid, Gel, Gel With Additives, or Gaseous for Any Concept

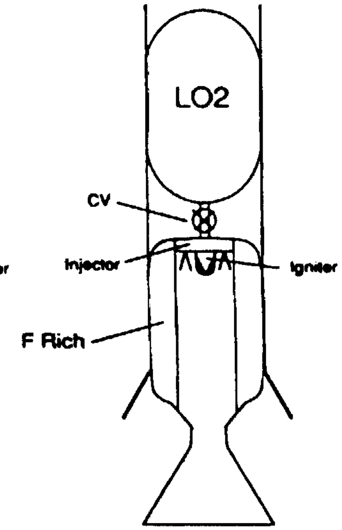
These Are the Six Combustion Schemes, Shown Schematically. Concepts A and C Use Pure Fuel Solid Grains. Others Contain Some Oxidizer, and Grain F Contains All of the Oxidizer in the Grain

HYBRID CONCEPTS

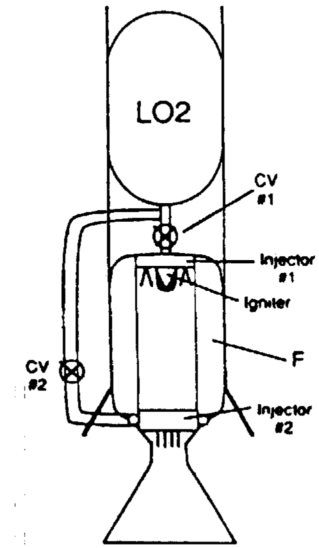
CONCEPT: A
(Conventional Hybrid)



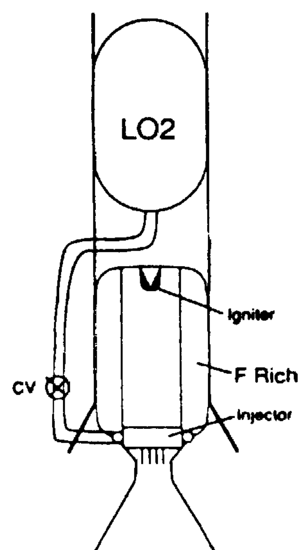
CONCEPT: B
(Quasi-Hybrid)



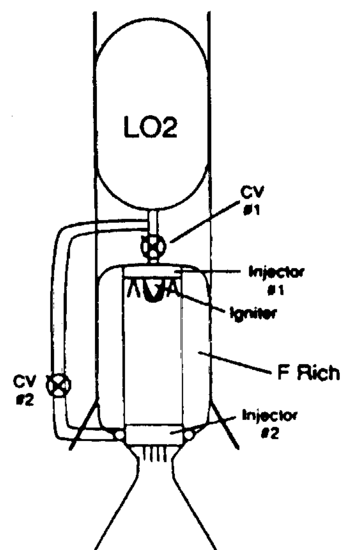
CONCEPT: C
(Hybrid-Liquid Staged Combustion)



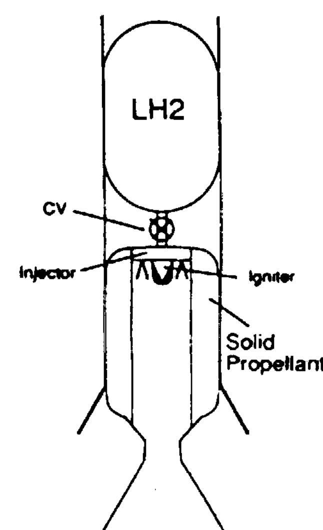
CONCEPT: D
(Solid-Liquid Staged Combustion)



CONCEPT: E
(Quasi-Hybrid-Liquid Staged Combustion)



CONCEPT: F
(H2 Augmented Solid)



This Is Our Combustion Scheme Screening Chart. As Most Others in This Study, It Summarized Our Screening Criteria at the Left and Our Concept Candidates Across the Top. We Added Concept G, Which Is Similar to F, Except High Density, Non-Cryogenic MMH (a Typical High Performance Amine Fuel) Was Substituted for the H_2 of Concept F. However, F and G Were Screened Out Because of the Operational Need for Extinguishment. Concept C Is a Staged Combustion Concept Having a Conventional Hybrid Gas Generator Ahead of the Main LO_2 Injector. This Two-Injector Concept Has No Oxidizer in the Fuel Grain, and the Gas Generated Must Be Fuel-Rich. The Technology for Low Mixture Ratio, Low Temperature, Fuel-Rich Hybrids Is Not Available and May Never Be, Since Pure Hybrids Depend Upon Heat From the Combustion to Vaporize the Fuel

Four Concepts Passed the Screen (All With Comments) A, B, D, and E: Two Single-Stage and Two Two-Stage Combustion Concepts, Three Having Fuel-Rich Solid Propellant

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LEVEL 2—COMBUSTION METHOD CONCEPT SCREENING

Screening Criteria	A Conventional Hybrid	B Quasi-Hybrid	C Hybrid-Liquid Staged Combustion	D Solid-Liquid Staged Combustion	E Quasi Hybrid-Liquid Staged Combustion	F H_2 Aug. Solid	G MMH Aug. Solid
I. Safety A. Explosive B. Ignition C. Environment and Toxic D. FMEA 1 E. Physical	(Avoid Injector Clogging and Overheating)					No LH_2 Temp	
II. Availability			No— Low MR Hybrid				
A. Technology B. Development C. Production D. Maintenance E. Reliability	Poor ERE Poor ηI_{sp}	Reduced I_{sp} Theo		Reduced I_{sp} Theo	Reduced I_{sp} Theo		
III. Operability A. Geometric						No— HRB Volume	Lower Isp
B. Operation						No— Extinguishment	No— Extinguishment
Screening Result	Yes	Yes	No	Yes	Yes	No	No

LEVEL 3
SUBSYSTEM OPTIONS

This Concept Generation Matrix Showed Several Ways to Pressurize the LO_2 Tank, for Pump or Pressure Fed HRBs Having Bleed or Topping Cycles. In Addition to the Four Generic Types of Pressurization Possible Considering Flow Scheme and Equipment Type, the Gas Source Type Was Also Considered: Stored Gas, Boiled Liquid (Autogenous), and Gasified Solid (Azide)

LEVEL 3—OXIDIZER TANK PRESSURIZATION CHART

Four Generic Types of Pressurization for Oxidizer Are Possible

	Pumped	Pressurized
Topping	A. Solid Gas Generator Flows Through Turbine Into Combustor a. Autogenous Tank Pressurization b. He Tank Pressurization	C. Gas Pressurant a. He b. Heat Gas Without Dump c. Pulsed Azide Gas Generator + Pressure Switch
Bleed	B. Solid Gas Generator Bleed Flows Through Turbine and Overboard a. Autogenous Tank Pressurization b. He Tank Pressurization	D. a. Steady Flow Azide Gas Generator + Overboard Dump Regulator b. Steady Flow, Gas Generator Flow Heats Regulated Gas, + Overboard Dump

The Next Two Sheets Schematically Illustrate the Concepts Generated. A Ninth Concept Is to Preheat the LO_2 So It Self-Pressurizes Throughout the Mission (VAPAK). A Tenth Concept for Tank Pressurization Is to Bleed GO_2 Turbine Exhaust to the Tank. Neither Required Schematic Representation

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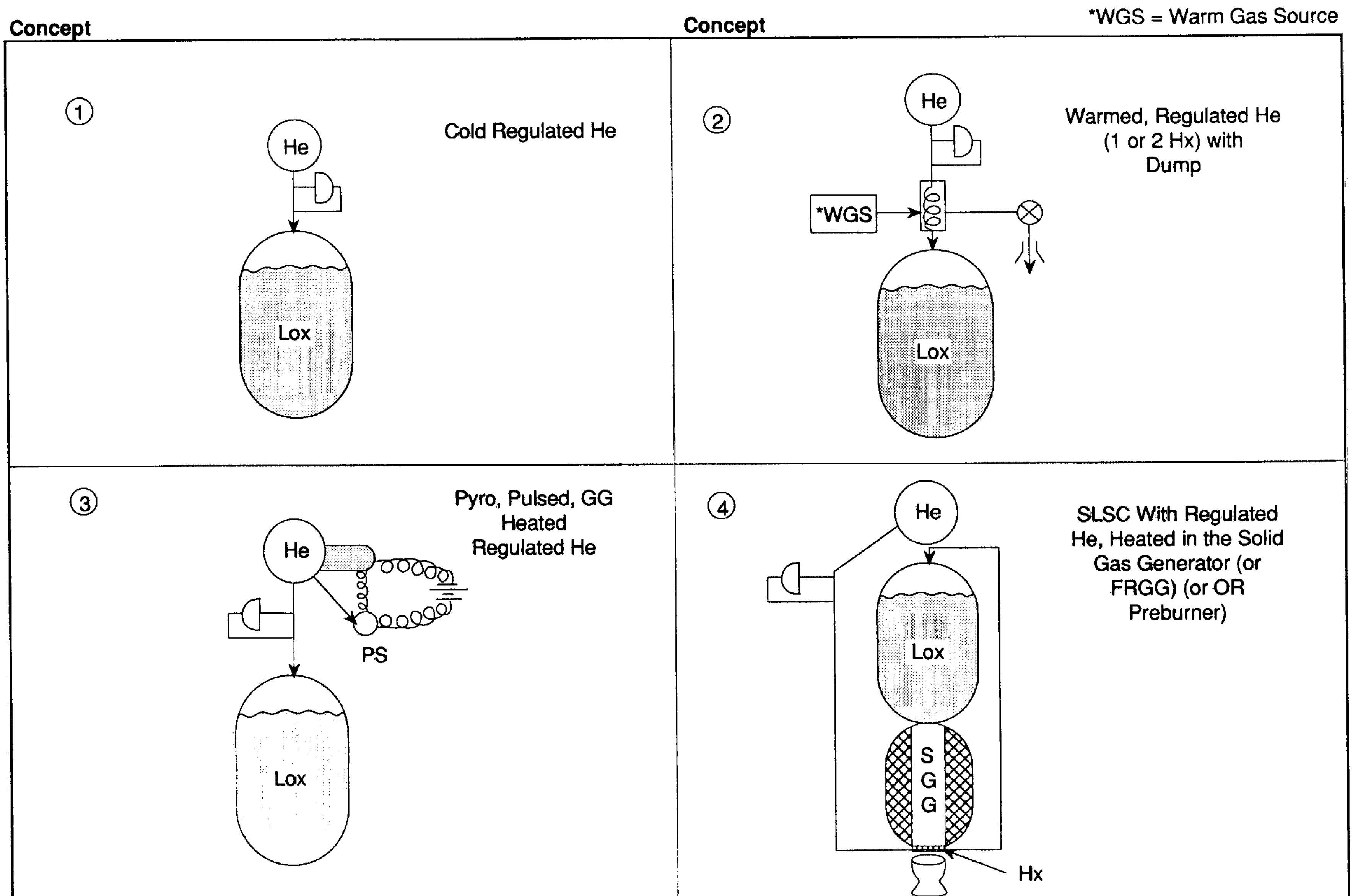
HRB TANK PRESSURIZATION OPTIONS



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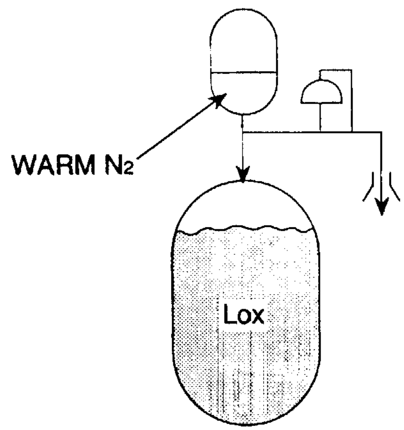
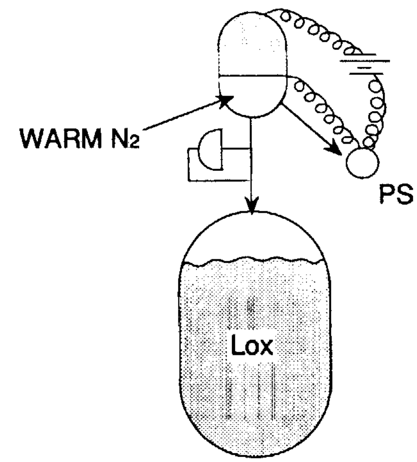
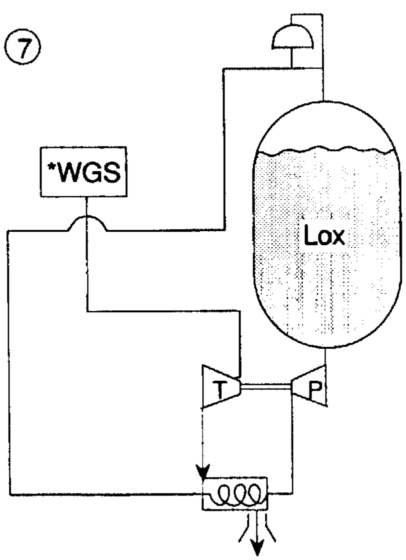
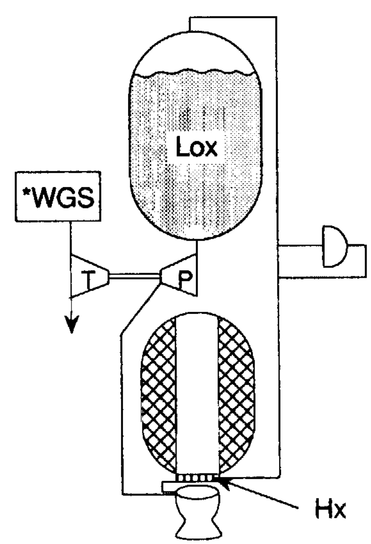
HRB TANK PRESSURIZATION OPTIONS (CONT)

*WGS = Warm Gas Source

Concept	Concept
⑤  Azide GG with Overboard Dump Regulator	⑥  Wafer Pulse Azide GG with Regulated N_2
⑦  Autogenous, with Dumped Warm Gas Heating	⑧  Autogenous with Lox Heated in the Solid Gas Generator or Oxid. Rich Preburner

This Screening Chart Eliminated Four LO_2 Tank Pressurization Concepts for Both Pump and Pressure Fed HRBs, and a Fifth for Pressure Fed Application. The VAPAK Concept Requires Facility Heating; It Is Screened Out Because We Believe LO_2 Ice Will Form and the Tank Cannot Be Emptied Into the Pump (Not a Pressure-Fed Concept), Because of Large Temperature and Pressure Reductions in the Tank Near the End of Its Action Time. The Azide Concepts (5 and 6) Are Excessively Heavy, and They Require Either Large Cryogenic Bladders (Technology Issue) or There Is a Safety Problem With Li or Na in the LO_2 Tank. Pyrotechnically Heated He Reduces the He Bottle Size, but Not Its Weight and a Heavy HRB for Pressure Fed Use. More Importantly, the Fuel-Rich Pyrotechnic Products Should React in the LO_2 Tank, or Require a Bladder. Finally, Cold He for Pressure Fed HRBs Violates the "No Large HP Bottle" Safety Criterion and Size and Weight Limits for HRBs.

This Leaves Only Heated He for Pressure Fed Pressurization - the Same Result We Had on Our Prior LRB for STS Study for NASA in 1988. Six Choices Remain for Turbopump Fed HRBs

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OXIDIZER TANK PRESSURIZATION SCREENING

Priority A—Level 3

	Regulated Helium (Near or Inside LO_2 Tank)				Heated H_2 From Aside		Autogenous Heated LO_2, TF Only			
Concept No.	1 Cold	2 With Warm Gas Dump Heating	3 With Pulsed Pyro Heating	4 With MX and GG Warm Gas (No Dump)	5 Bleed Regulator	6 Pulsed	7 Warm Gas Dump Heated LO_2	8 Warm Gas Heated NX	9 VAPAK	10 Cycle Bleed
Pressurization Cycle Type:	Topping	Bleed	Topping	Topping	Bleed	Topping	Bleed	Topping	Either	Bleed GO_2 From Turbine Exhaust
III. Safety										
A. Explosive										
B. Autoignition										
C. Toxicity D. Environment Catastrophic Failure Mode	No - Large, High Pressure Bottle (PF)		No - LO_2 Tank Explosion Potential	SSC NX Burnout in 3589 °K (6,000°F)	No -					
E. Physical			Either	High Pressure Gas So Use O_2- Rich Pre-burner	LO_2 Tank Explosion Potential	Either	LO_2-to-Fuel Rich Warm Gas HX (In Solid Case, or With Fuel Rich Bleed Flow)			
II. Availability A. Technology			No? - Cryogenic Bladder Technology		No? Cryogenic Bladder Technology					
B. Development Risk										
C. Producibility									SSC Needs Boost Pump	
D. Maintainability E. Reliability, Payload Availability		SSC Needs a Hot Gas Generator Subsystem					SSC Needs a Hot GG (Bleed) Subsystem	SSC Needs an O2-Rich Pre-burner for Heat		
III. Operability A. Geometric	No - Large, Heavy (PF) HRB		No - Large HRB		No - Heavy HRB	No - Heavy HRB				
B. Operational									No - Pressure Loss	
Screening Result:	No - PF Yes - TF	Yes (SLSC Favored)	No	Yes	No	No	Yes - (STS Uses This for ET Pressurization)	Yes	No	Yes

*SSC - Single Staged Combustion ** SLSC - Solid Liquid Staged Combustion
PF = Pressure Fed
TF = Turbopump Fed

This Concept Generation Matrix Identified Turbine Drive Gas Sources by Considering Drive Gas Chemistry Versus Its Physical Origin (Solid, Hybrid, Liquid). The Turbine Drive Gas Sink Is Assumed to Be a Dump Nozzle (Bleed Cycle) or the Main Combustor (Topping Cycle). Recombustion of Turbine Exhaust Gas at Low Pressure (Bottom Cycle) Was to Be Considered Later for Any Successful Candidates.

Eight of the 19 Candidates Are Obviously Not Feasible or Unnecessary as Indicated. Eleven Concepts (Plus Topping and Bleed Cycle Versions of Numbers 1 and 2) Are Shown Schematically on the Next Two Sheets

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HRB PUMPED OXIDIZER PRESSURIZATION TURBINE DRIVE CYCLES TOPPING OR BLEED CYCLE OPTIONS

Turbine Drive Gas Sink = Main Combustor or Bleed Nozzle
Turbine Drive Gas Source

Chemical Options \ Physical Options	Solid GG (SLSC) Main SLSC GG	Solid GG (Either Type) (Combustor) Separate GG	(SSC) Hybrid GG	(Either Type) (Combustor) Biprop GG
Fuel Rich (Either Cycle)	1 SLSC Motor Case	3 (FRGG)	~~SSC Motor Case Flow (=Too Hot)~~	~~LO_2 + CH_4 Fuel Rich Preburner (Too Much Fuel Required)~~ 7 LO_2 + CH_4 Fuel Rich GG
Fuel Rich With Diluent (Either Cycle)	2 SLSC Motor Case + Diluent	~~Diluent Unnecessary~~	6 SSC Motor Case Flow + Diluent	~~Diluent Unnecessary~~
Stoichiometric and Diluent (Bleed Only)	~~LO_2 + SLSC Motor Case Bleed + Diluent (Very High Case Pressure)~~	10 SSGG + Diluent	4 FRGG + LO_2 + Diluent ~~SSC Motor Case Bleed + LO_2 + Diluent (No Combustion Potential)~~	8 LO_2 + LHC GG + H_2O Diluent
Oxidizer Rich (Topping Only)	~~LO_2 Flow Motor Case Bleed (Very High Care Pressure)~~	11 FRGG + LO_2 Preburner	5 FRGG + LO_2 Preburner ~~LO_2 Flow + SSC Motor Case Bleed (No Turbine Drive Pressure)~~	9 LO_2/LHC Ox-Rich Preburner

SLSC = Solid/Liquid Staged Combustion
SSGG = StoichiSolid Gas Generator

SSC = Standard Single Combustor
FRGG = Fuel Rich Gas Generator
LHC = Liquid Hycrocarbon

MHD10/Hybrid/Pt 7

Topping Cycles Are Shown on the Left of This Two-Sheet Diagram, Bleed Cycles on the Right. Concept Numbers Correspond to the Concept Generation Matrix Numbers. Some Concepts Are Valid Only for Single Stage Combustion Schemes, and Some for Only the Solid/Liquid Staged Combustion Types. Those Applicable for Either Are So Marked, and the Version Shown Is Noted

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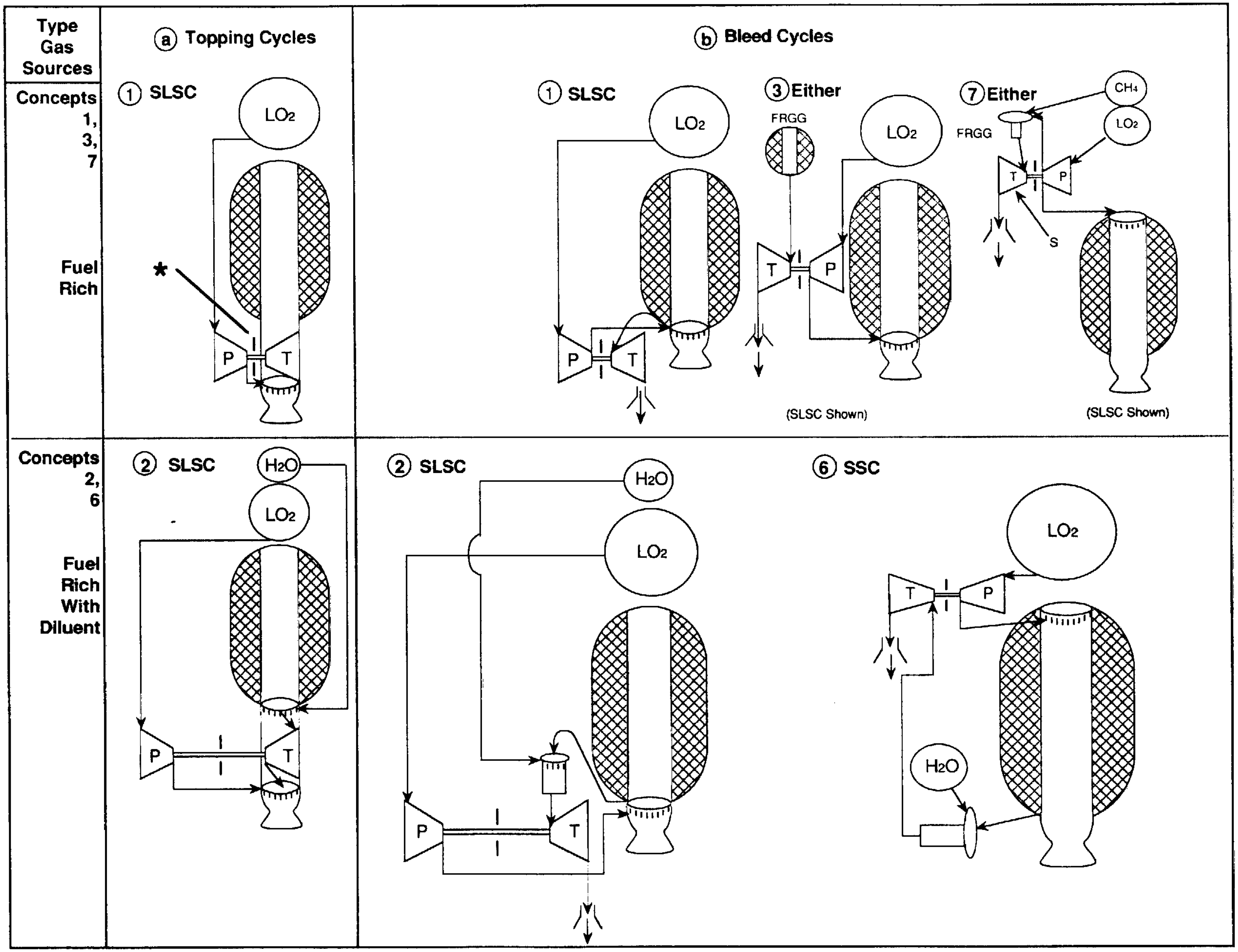
HRB PUMP FED TURBINE DRIVE CYCLE OPTIONS



S = Facility N_2 Bleed Start FRGG = Fuel Rich Gas Generator * Signifies Buffer Seal in Turbomachines

1.9.1.115

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HRB PUMP FED TURBINE DRIVE CYCLE OPTIONS (CONT)

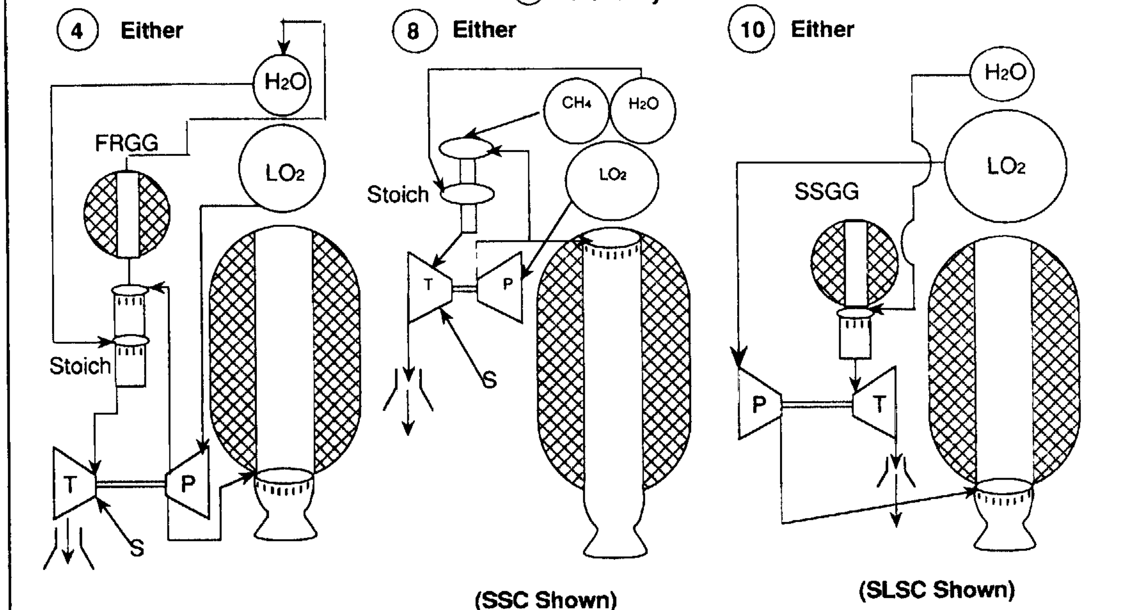
Concepts 4, 8, 10

Stoichio metric With Diluent

(a) Topping Cycles

No Topping Cycle With Diluent (Low I_{sp})

(b) Bleed Cycles

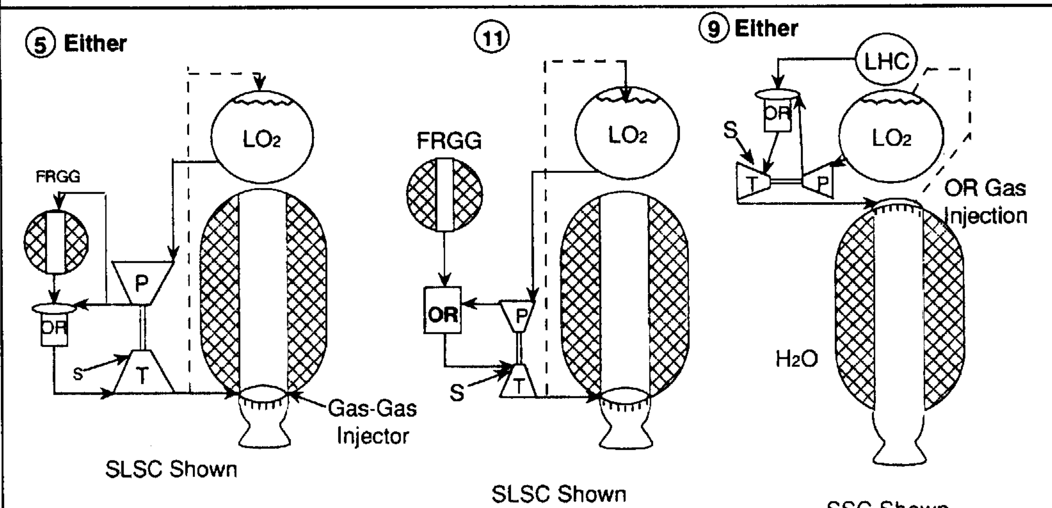


Type Gas Sources

Concepts 5, 9, 11

Oxid-Rich Pre-Burner

(a) Topping Cycles



(b) Bleed Cycles

No Oxidizer-Rich Bleed- (Safety and Low I_s) (Without the Bottom Cycles)

S = Facility N_2 Bleed Start SSGG = Stoich Solid Gas Generator LHC = Liquid Hydrocarbon Fuel OR = Oxidizer Rich FRGG = Fuel Rich Gas Generator 1.0.1.116

This Screening Chart Eliminated Seven Turbine Drive Cycles. All Have Fuel-Rich Turbine Drive Gas Incompatible With LO_2 Being Pumped. Item 1 of FMEA Chart (Page 19) Recommends Assured Buffer Seal Operation or No Fuel-Rich Gas in TPA. Since We Cannot Assure Seal Operation (and Can Realize Cost and Safety Improvements by Eliminating the Seal and Its Supply and Feed System) We Elected to Eliminate Buffer Seals. Cycles 1 and 2 Had Other Negatives as Well, Including Obvious Development Risk via TPA/Solid Case Interactions. This Leaves Us With Six Cycle Candidates and an Incomplete Risk Assessment

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TURBOPUMP FED TURBINE DRIVE GAS CYCLE SCREENING

PRIORITY A LEVEL 3	TOPPING CYCLES					BLEED CYCLES							
	1A	2A	5A	9A	11A	1B	2B	3B	4B	6B	7B	8B	10B
COMBUSTOR TYPE:	SLSC	SLSC	EITHER	EITHER	EITHER	SLSC	SLSC	EITHER	EITHER	SSC	EITHER	EITHER	EITHER
I. SAFETY													
A. EXPLOSIVE													
B. AUTO-IGNITION													
C. TOXICITY ENVIRO	-NO-	-NO-				-NO-	-NO-	-NO-		-NO-	-NO-		
D. CATASTROPHIC	TURB	TURB				BUFFER	BUFFER	BUFFER		BUFFER	BUFFER		
MODES OF FAILURE	PLUGS	PLUGS				SEAL	SEAL	SEAL		SEAL	SEAL		
E. PHYSICAL	-NO-	-NO-											
II. AVAILABILITY	BUFFER	BUFFER											
A. TECHNOLOGY	SEAL	SEAL											
ACQUISITION DATE													
B. DEVELOPEMENT	-NO-	-NO-											
RISK	*TC	TC											
C. PRODUCIBILTY													
D. MAINTAINABILITY													
E. RELIABILITY, PAYLOAD, AVAILABILITY													
III. OPERABILITY													
A. GEOMETRIC (Isp, MF, rho)													
B. OPERATIONAL													
SCREENING RESULT:	NO	NO	YES	YES	YES	NO	NO	NO	YES	NO	NO	YES	YES

*TC=TOPPING CYCLE DEVELOPEMENT COST/RISK

We Assessed the Relative Risk of 13 Components Potentially Used in the Five Surviving Candidate Turbine Drive Cycles. Risks Included All Availability Category Elements

LEVEL 1, 2, AND 3 RISK CHART

No.	Risk	Category	Penalty	Combustion Method Notes
0	Large Fwd Injector	Development	1/2	A, B Only
1	Propellant Development	Tech. Acq.	1	One of These,
2	Active Radial Flow Injector	Development	1	D and E Only
3	Hot Bleed Port	Development	1	
4	Oxidizer Rich Preburner	Development	1/2	
5	Topping Cycles	Development	1/2-2*	
6	Second Fluid or Gas Generator	Reliability	1/2	
7	Bleed Loss	Payload	1/2	
8	Sliver Loss	Payload	1/2-1**	A, B Only
9	Mixing (MRD) Loss	Payload	1/2-1**	A, B Only
10	Quasi-Hybrid Theo I_s Loss	Payload	1/2	B, D, E Only
11	ΔP Injector	Payload	1/2	D, E Only
12	GO_2 Turbine (Low Temperature)	Development	1/2	
13	Boost Pump or Tk. Press. Sys.	Reliability	0-1/2***	A, B Only

* Depends on Degree of Development Dependency Between TPA and TCA
** Quasi-Hybrid Losses Should Be Less
*** 0 for Ox-Rich Turbine Drive Cycles and 1/2 for Others

Appropriate Component Risks Were Summed for Each Surviving Candidate Turbine Drive Cycle for Use in Both Single Stage Combustors and Solid/Liquid Staged Combustion HRBs. The Totals Showed That Cycles 10b (Bleed) and 11a (Topping) Have the Lowest Risk, All Others Scoring About Twice as Great as the Best Situation (Bleed Cycle With an SLSC HRB). Cycles 10b and 11a Survived the Screening.

C-2

At This Point the Bottoming Cycle Option Was Added, and It Became Obvious That Burning Off the Exhaust of an Oxidizer-Rich (Cool) Turbine With the TCA Fuel-Rich Boundary Layer Would Also Have a Risk Score of 1, the Diluent System of Cycle 10b Could Be Eliminated, and So Could Its Autogenous Heat Exchanger. Thus, the Bleed Burnoff Cycle Was Added to the Candidate List

PUMP FED TURBINE DRIVE CYCLE RATING AND SCREENING CHART

Level 3

Risk No.	Candidate Concepts 4b	5a	8b	9a	10b	11a
3	--	--	--	--	--	--
4	--	1/2	--	1/2	--	1/2
5	--	1/2	--	1/2	--	1/2
6	1	1/2	2	1/2	1/2	--
7	1/2	--	1/2	--	1/2	--
12	--	1/2	--	1/2	--	1/2
13	0 to 1/2	--	0 to 1/2	--	0 to 1/2	--
Totals	2 (SSC) 1-1/2 (SLSC)	2 (Both)	3 (SSC) 2-1/2 (SLSC)	2 (Both)	1-1/2 (SSC) 1 (SLSC)	1-1/2 (Both)

Score	SLSC	SSC
1	10b	--
1-1/2	11a	10b, 11a
Risk Screen		
2	5a, 9a	4b, 5a, 9a
2-1/2-3	8b	8b

Three Cycles Remain to Be Investigated With FOM:
10b, 11a, and a Bottoming Cycle Version of the Oxidizer-Rich Bleed Cycle (With Nozzle Burnoff)

The Final Turbine Drive Cycle Candidates Are Pictured Below, Along With a Five Feature Qualitative Assessment (G = Good, B = Bad, N = Neutral Feature) at the Bottom of Each Schematic.

The Bleed Burnoff Cycle Appears to Have the Best Good-Bad Relationship of the Three Cycles and Appears Smaller and Simpler

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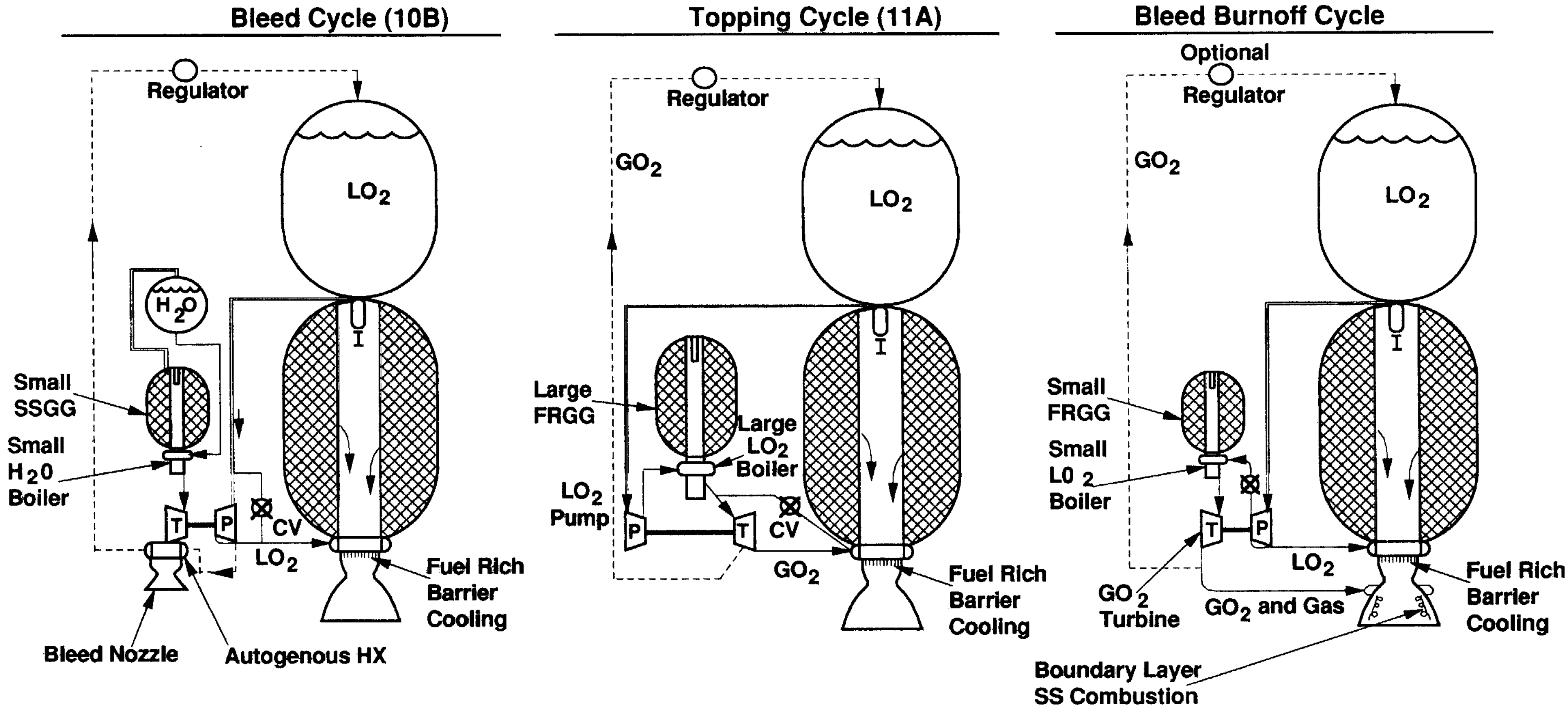
HRB FOR STS
FINAL TURBINE DRIVE CYCLE CANDIDATES



Bleed Cycle (10B)	Topping Cycle (11A)	Bleed Burnoff Cycle
G - Neutral Gas Turbine	N - Cool Oxidizer Rich Turbine	N - Cool Oxidizer Rich Turbine
G - Small Components	B - Large Components	G - Small Components
B - Bleed Loss	G - No Bleed Loss	G - No Bleed Loss
B - Barrier Cooling Loss	B - Barrier Cooling Loss	G - No Barrier Cooling Loss
B - Autogenous HX	B - Gas/Gas Injector	B - Small C_F and Comb. Loss

SSGG = Stoichiometric Solid Gas Generator
FRGG = Fuel - Rich Solid Gas Generator

3.16.0.11

This Chart Shows That There Were Acceptable Tank Pressurization Methods for Each Feed System Remaining. Each Turbopump Fed System Has Three Choices, but the Pressure Fed Has Only One. Dump Heated Helium LO_2 Tank Pressurization Is Selected for the Pressure Fed HRB

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DRIVE CYCLE/TANK PRESSURIZATION COMBINATION OPTIONS

HRB Tank Pressurization Options	LO_2 Feed Cycle Options			
	Bleed (10b)	Topping (11a)	BBC	Pressure Fed
1 Cold He	Yes	Yes	Yes	No— Safety, Size
2 Dump Heated He	Yes	No— Not Necessary	No— Not Necessary	Yes
4 No Dump Heated He	No— HX Develop.	Yes	Yes	No— HX Develop.
7 Autogenous, Dump Heated	Yes	No— Not Necessary	No— Not Necessary	No Perf, Complex
8 Autogenous, No Dump Heated	No— Cat. 1 Failure Mode	No— Not Necessary	No— Not Necessary	No— Perf, Complex
10 GO_2 Cycle Bleed	No— Not Available	Yes	Yes	No— Not Available

Legend: No Option Screened Out
Yes Option Retained

Conclusions:
1. 10 of 24 Combinations Retained
2. Each TF Cycle Has 3 Options
3. PF Has One Option-Heated He (Dump)
4. No Dump Autogenous Pressurization Is Screened Out

We Postulated Five Kinds of Expansion Nozzle Types for HRB Use, Including:

- DeLaval
- DeLaval With Extendible Exit Cone (SLEEC)
- Plug Nozzle (Annular Throat + Center Body)
- Plug Cluster (Circular Array of Canted and Scarfed Nozzles)
- Expansion-Deflection (Shortened Nozzle Contour With Multiple, Non-Axial Throats Near Its Center Point)

The E-D and Plug Type Nozzles Were Screened Out Because of Potential Center Body Failures That Result in Solid Case Overpressure and Rupture. The Applicability of the Plug Cluster Depended Entirely on How Many Nozzles Appeared Optimum. The Conventional DeLaval and Its Potential Exit Cone Extension Passed the Screen

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THRUST CHAMBER ASSEMBLY (TCA) OR NOZZLE TYPE SCREENING

Priority A—Level 3

Class of TCA	DeLaval	Plug	E-D	Plug Cluster	SLEEC
Considerations					
I. Safety					
A. Explosive					
B. Autoignition					
C. Toxicity and Environmental					
D. Catastrophic Failure Modes		No Centerbody Failure	No Centerbody Failure		
E. Physical					
II. Availability					
A. Technology Acquisition Date		New Configuration	New Configuration	New Configuration	
B. Development Risk					
C. Producibility					
D. Maintainability		Throat Growth	Throat Growth	Throat Growth	
E. Reliability, Payload Availability					
III. Operability					
A. Geometric		No Mass Fraction			
B. Operational				Throat Cooling	
Screening Result:	Yes	No	No	?	Yes

*Nil I_{sp} Difference Among All Candidates If Done Well, Except for Throat Growth Impact (SLEEC Excluded; If It Can Extend Out to Near the Aft Skirt Diameter It Can Enhance)

The Next Two Sheets Show the Logical HRB Geometry Options. Configurations B and A, Were Studied in Accordance With the Contract SOW. B Is the Small HRB Scenario, and A Is the Baseline Large HRB Scenario. Configurations C, D, E, F Are Other Large HRB Options, of Which C and D Are Viable, and E and F Are Not

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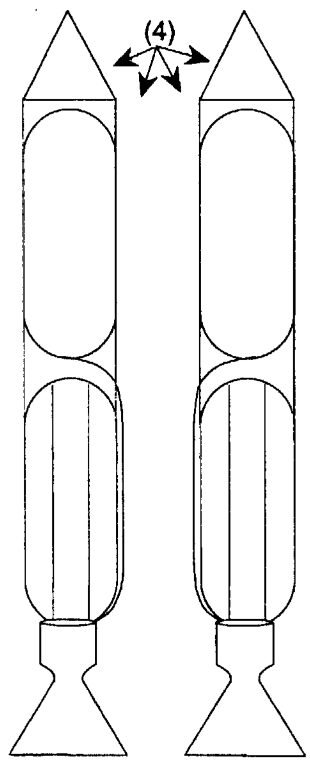
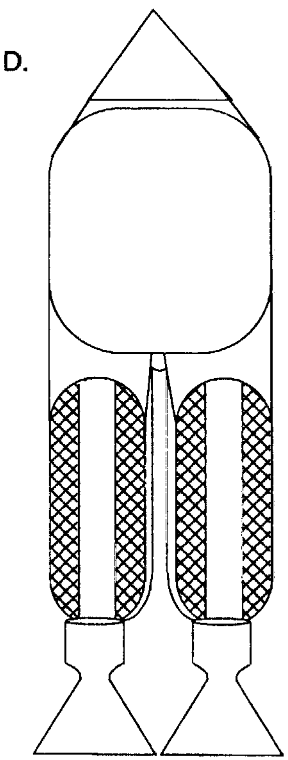
TCA,TANKAGE, AND MOTOR CASE OPTIONS

B. Separate HRBS (SOW Option)



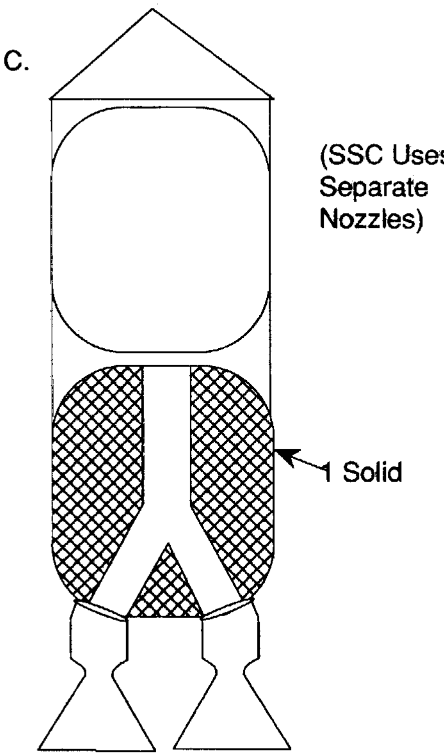
(Vehicle Flexibility)
Small TCAS
Cases, Nozzles
Low Development Cost

D. Separate Solids (Martin LSAT Option)



(Uses Separate HRB TCAs and GGS)

C. Separate TCAs (Aerojet Option)



(Uses Separate HRB TCAs
And Nozzles)

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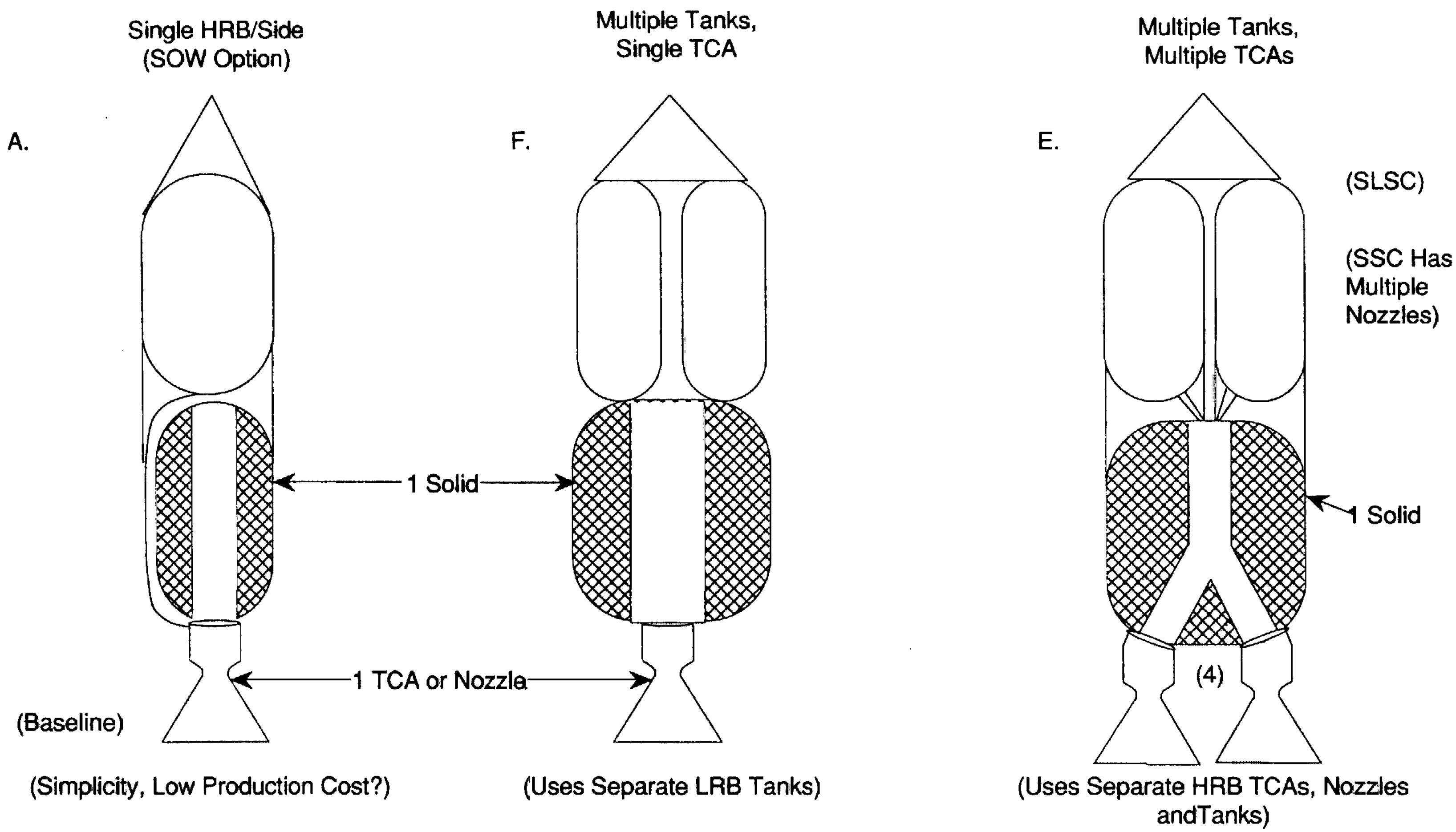
TCA,TANKAGE, AND MOTOR CASE OPTIONS (CONT.)



1.9.1.111

This Screening Chart Selected All Single LO_2 Tank Versions of the Large HRB. It Identified Differences in TCA-Out Capabilities of the Three

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NUMBER OF TCAs (OR NOZZLES) AND MOTOR CASES SCREENING

Priority A — Level 3

Options	One TCA or Nozzle /One Motor Case	Four* Chambers or Nozzles /Four Motor Cases	Four Chambers or Nozzles /One Motor Case
I. Safety			
A. Explosive			
B. Auto Ignition			
C. Toxicity, Environmental			
D. Catastrophic Failure Modes			
E. Physical			
II. Availability			
A. Technology Acquisition Date			
B. Development Risk	TCA Size Outside of SOA		
C. Producibility			
D. Maintainability			
E. Reliability, Payload Availability	No TCA-Out Capability	TCA Out Reliability Is Provided for RTLS Abort or Abort to Canary Is.	TCA-Out Reliability Enhancement if SLSC GG Flow Control Is Supplied, But Control Valve Closure Is a Critical Factor
III. Operability		Reduce Length SSC Loses Isp SLSC Gains Isp	Reduce Length SSC Loses Isp SLSC Gains Isp
A. Geometric			
B. Operational			
Screening Result	Yes	Yes	Yes

* See Cost Study for LRB, as Well as Use Flexibility, Per SOW Use Scenario Options

There Are at Least Three Overall Shapes That Have Been Used in American and Soviet Strap-on Boosters and Appear Feasible for HRBs. They Feature Cylindrical and Conical Liquid Tanks and Solid Cases

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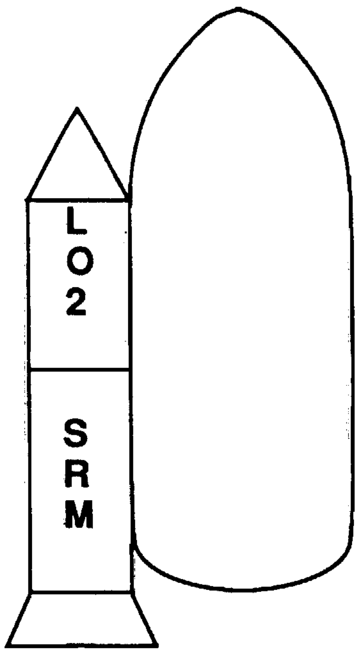
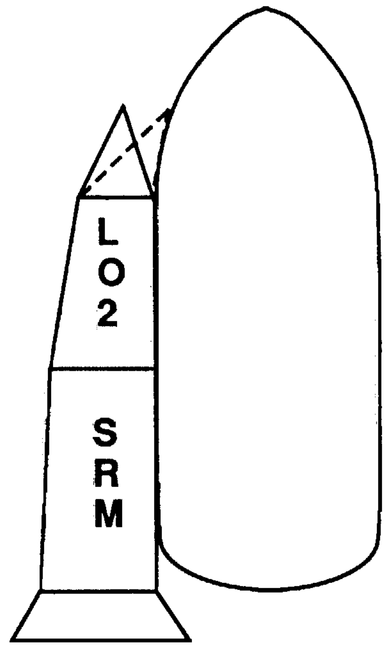
HRB SHAPE OPTIONS



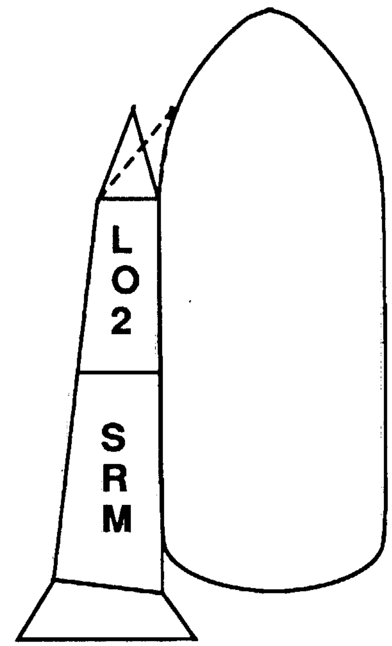
Cylindrical Motor
Case and LO_2 Tank

Standard
Concept



Cylindrical Motor
Case With Tapered
and Canted LO_2 Tank

Reduced Wave
Drag Loss



Tapered Motor
Case and LO_2 Tank
With Canted Skirt

Greater Reduction in
Wave Drag Loss

3.15.0.20

Our Liquid Propellant Tank Screening Recommended Study of Both Cylindrical and Conic Shapes. It Recalls, However, That Composite SRB Motor Case Tests Showed Insufficient Bending Stiffness, So Only Metallic and Composite Overwrapped Tanks Were Investigated (Only Pump Fed Can Use Metal Tanks Without High Strength Hoop Overwrap)

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LIQUID PROPELLANT TANK SCREENING

Priority A—Level 3

Options:	Type of Construction: Metallic	Type of Construction: Composite Overwrapped Metal	Type of Construction: Composite	Shape: Cylindrical	Shape: Tapered
I. Safety					
A. Explosive					
B. Autoignition					
C. Toxicity and Environmental					
D. Catastrophic Failure Modes					
E. Physical					
II. Availability					
A. Technology Acquisition Date					
B. Development Risk					
C. Producibility					
D. Maintainability					
E. Reliability, Payload Availability					
III. Operability					
A. Geometric	No—Weight, Size (PF)			Baseline	Improved Aerodynamics
B. Operational			No—Stiffness Criteria		
Screening Result:	TF = Yes (Al-Li) PF = No	TF = Yes (Al-Li) PF = Yes	No	Yes	Yes

TF = Turbopump Fed
PF = Pressure Fed

M4D10/Hybrid/P1 6

Our Liquid Feed System Screening Showed That:

- All Feed Lines Should Be External to the Solid Case (Inspection, Insulation, Support, and Volumetric Reasons Abound)

- Pressure Fed Must Use Pressurized Lines

- An Aft Mounted Turbopump Requires Suction Lines to Avoid the Need for a Boost Pump, Due to the Head Applied to the LO_2 by the Length of the Solid Case and HRB Acceleration

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LIQUID FEED SYSTEM SCREENING (SLSC ONLY)

Priority C — Level 3

	Type		Location	
Feed System Options	**Pressure Lines**	**Suction Lines**	**External Lines**	**Internal Lines**
I. Safety	(Fwd.Mounted) (TPA or PF)	(Aft Mounted TPA, Pumped Only)		
A. Explosive				
B. Auto Ignition				
C. Toxicity, Environmental				
D. Catastrophic Failure Modes				
E. Physical				
II. Availability				
A. Technology Acquisition Date				
B. Development Risk				No - Inspection Insulation On Line
C. Producibility				
D. Maintainability				
E. Reliability, Payload Availability	Boost Pump or Tk. Pressure Sys. Req'd			
III. Operability				
A. Geometric	More Tank Pressure Required			Solid Motor Void Fraction?
B. Operational				
Screening Result	PF = Yes TF = No	Yes	Yes	No

* TF = Turbopump Fed
PF = Pressure Fed

Four Types of TVC Were Considered for HRBs

TYPES OF TVC APPLICABLE TO SLSC HRB FOR STS

Priority B—Level 3

Type/Variation	Reason/Application/Problem
I. Moveable Exit Cone a. Above the Throat b. At the Throat c. Below the Throat d. Flex Seal e. Ball Seal	Available for SRM/Actuator Power, Fluid? Power Takeoff From TPA? Hydraulic System? Electric Actuators?
II. Fixed Nozzle With Four Fluid Injection Points a. LITVC 1. Reactive - LO_2 2. Nonreactive b. GITVC 1. GG Bleed 2. Other Source (TCA) c. Bipropellant TVC 1. GG Bleed + LO_2 + 4 Injectors	Effective Gimbal Angle?
III. Multiple TCAs With On/Off Control a. Four TCAs b. Eight to Ten TCAs c. Four Nozzle Rectangular TCAs + Short Plug Nozzle	Sufficient Authority and Angle? TCA Out Interaction? Baseline? TCA Out Capability? Reaction Rate? Improve P/Y Moment Arm?
IV. Head End Gimbal	• Large Diameter, High Pressure, Warm, Leakfree Joint Required • Large Gimbal Power Required - Heavy Mass, Long Moment Arm, Friction and Hysteresis

Two of the Four Were Not Feasible for NASA HRBs. Multiple TCA Throttling Could Not Supply Sufficient Yaw Moment, and Head End Gimbaling Invites a Catastrophic Warm Gas Leak or Is Not Applicable to the Design. This Leaves the Movable Exit Cone TVC With a Questionable LITVC Option Using LO_2. This Has Never Been Demonstrated and May Not Result in Sufficient Side Force

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Solid Propulsion

TVC SCREENING

Priority B — Level 3

	Movable Exit Cone	LI/GI/BI TVC (LO2 Injection Favored)	Multiple TCA/Motor Throttling (Large SLSC or Small HRB)	Head End Gimbal (SLSC Only)
I. Safety				
A. Explosive				
B. Auto Ignition				
C. Toxicity, Environmental				
D. Catastrophic Failure Modes			Large HRB = No (Loss of Control With Large Motor Out)	No - Warm Gas Leak
E. Physical				
II. Availability				
A. Technology Acquisition Date				High Pressure Bellows or Hot, Zero Leak Large Size Seal
B. Development Risk		?		
C. Producibility				
D. Maintainability				
E. Reliability, Payload Availability	Actuators	Valves	Valves	Actuators
III. Operability				
A. Geometric				
B. Operational		Side Force?	No - Yaw Moment Force Only	
Screening Results	Yes SRM Technology	Yes? Titan Technology + LO_2 Technology	No	No

We Reviewed Four Methods of Thrust Chamber and/or Nozzle Cooling for HRBs, and Screened Out Those Requiring LO_2 (or Second Fluid) Injection. Thus, Transpiration and Film Cooling Were Eliminated. However, Ablative, Fuel Rich Gas Barrier, and LO_2 Regenerative Cooling Were Retained. Regenerative LO_2 Cooling Was Applicable Only for Turbopump Fed Systems, and Smaller TCA or Nozzles, Because of the LO_2 Pressure Losses in the Cooling Jackets and the Related Costs Added to the Tankage or TCAs

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THRUST CHAMBER COOLING METHOD SCREENING (SLSC ONLY)

Priority B — Level 3

Options	Regenerative	Film/Barrier	Ablative	Transpiration
I. Safety	(Turbopump Fed Only)		(With or Without Gas Barrier)	
A. Explosive B. Auto Ignition C. Toxicity, Environmental D. Catastrophic Failure Modes E. Physical				
II. Availability A. Technology Acquisition Date B. Development Risk C. Producibility D. Maintainability E. Reliability, Payload Availability	Multiple TCAs Only	Film = No (Added Coolant System - LO_2 Excluded)	? Segmented Nozzle or Multiple TCAs	No - (Need Added Coolant System - LO_2 Excluded)
III. Operability A. Geometric B. Operational				
Screening Results	Yes = Pumped, Multi TCA Only No = Others	Film = No, Barrier = Yes (SLSCGG Gas)	Yes (With Barrier As Required)	No

There Is No Reason to Pre-Screen Out Combustion Stability Enhancement Methods

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COMBUSTION STABILITY AIDS

Priority B — Level 3

Options	Combustion Stability Aids: Delta P	Oxidizer	Orifice	Baffles	Acoustic Cavities	Aluminum or Boron
I. Safety						
A. Explosive						
B. Auto Ignition						
C. Toxicity, Environmental						
D. Catastrophic Failure Modes						
E. Physical						
II. Availability						
A. Technology Acquisition Date						
B. Development Risk	Liquid, HF & LF Interaction	Solid	Liquid HF & LF	Liquid HF	Liquid & Solid HF	Solid
C. Producibility						
D. Maintainability						
E. Reliability, Payload Availability						
III. Operability						
A. Geometric						
B. Operational						
Screening Results	Yes	Yes	Yes	Yes	Yes	Yes

HF = High Frequency Instability (Screaming)
LF = Low Frequency Instability (Chugging)

We Screened Solid Case Shapes and Did Liquid Tanks and Evaluated Cylindrical and Conical Shapes. Similarly, We Screened Composite Case Construction Out, Because of Its Demonstrated Lack of Stiffness. This Left Us With Metallic Motor Cases (for Pressure Fed HRBs) or Metallic With High Strength Composite Hoop Overwrap for the Likely Higher Pressure Turbopump Fed HRBs

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SOLID PROPELLANT CASE

Priority A — Level 3

Options	Type of Construction			Shape	
	Metallic	Metallic With Hoop Composite Overwrap	Composite	Cylindrical	Tapered
I. Safety					
A. Explosive					
B. Auto Ignition					
C. Toxicity, Environmental					
D. Catastrophic Failure Modes					
E. Physical					
II. Availability					
A. Technology Acquisition Date					
B. Development Risk					
C. Producibility					
D. Maintainability					
E. Reliability, Payload Availability					
III. Operability					
A. Geometric			No - Stiffness Criteria	Baseline	Improved Aerodynamics
B. Operational					
Screening Results	Yes	Yes For $P \gtrsim 10.34$ MPa (1,500 psia)	No	Yes	Yes

Grain Design Types Suggested Were the Modern, Internal Burning, Multiperforate Variety Used Often With Hybrid Motors, and Internal/External Used for Smaller Solid Motors Several Years Ago. The Latter Was Screened Out, Because the Center Grain Aft Support Failure Consequences Are Too Severe

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GRAIN DESIGNS

Level 3

Considerations	Grain Design Concepts		
	Internal Burn	Internal/External	Multiperforate
A. FMEA		Center Grain Aft Support Failure Mode - No	
B. Energy Efficiencies			
1. Sliver Losses	Small	Small	Large
2. Energy Release Efficiency, MRD Losses (Single Combustor Only)		Size, Design and Technology Dependent	
C. Mass Fraction Factors			
1. Volumetric Loading Fraction	Max	Less	Less
2. Insulation Requirements	Min	More	More
D. Operational Requirements			
1. Web Thickness	Less Than R	Less Than R/2	Min
2. Burn Area vs Time	Variable (By Design)	~Constant	Variable (Increases)
Screening Results			
SLSC, 4 x (1/4 Size Motor)	Yes	No	D = No , E = Yes
SLSC, Full Size Motor	Yes	No	D = No , E = Yes
SSC, 1/4 Size Motor	Yes	No	Yes
SSC, Full Size Motor	Yes?	No	Yes

Our Igniter Screening Suggests That We Should Use Forward-Mounted Igniters for HRBs. Surface Igniters Have Not Been Demonstrated With HRBs and Have No Redundancy Feature. Aft End Igniters Could Ignite the Aft End of a Hard-to-Light Grain, Compress the Air In Front of It, and Blow the Solid Case Out Again a Second Later

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IGNITER (REDUNDANT)

Priority C—Level 3

Class of Igniter	Location: Forward End	Location: Aft End	Location: Surface	Type: Pyrotechnic	Type: Hypergolic
Considerations					
I. Safety					
A. Explosive					
B. Autoignition					?
C. Toxicity and Environmental					
D. Catastrophic Failure Modes					
E. Physical					
II. Availability					
A. Technology Acquisition Date					
B. Development Risk		No—Not Demo'd With Hybrid			
C. Producibility	Interstage Access? 0.6 m (2-Ft.) Crawl Space—Pyro				
D. Maintainability					
E. Reliability, Payload Availability			No—(Redundancy)		
III. Operability					
A. Geometric		?			
B. Operational					
Screening Result:	Yes	No	No	Yes	Yes

PAST EXPERIENCE AND ONGOING STUDIES

- Solid Propulsion
- Liquid Propulsion

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SOLID PROPULSION EXPERIENCE

- Large Motors
- Case Design
- Fuel-Rich Propellant
- Grain Design
- Ignition Systems

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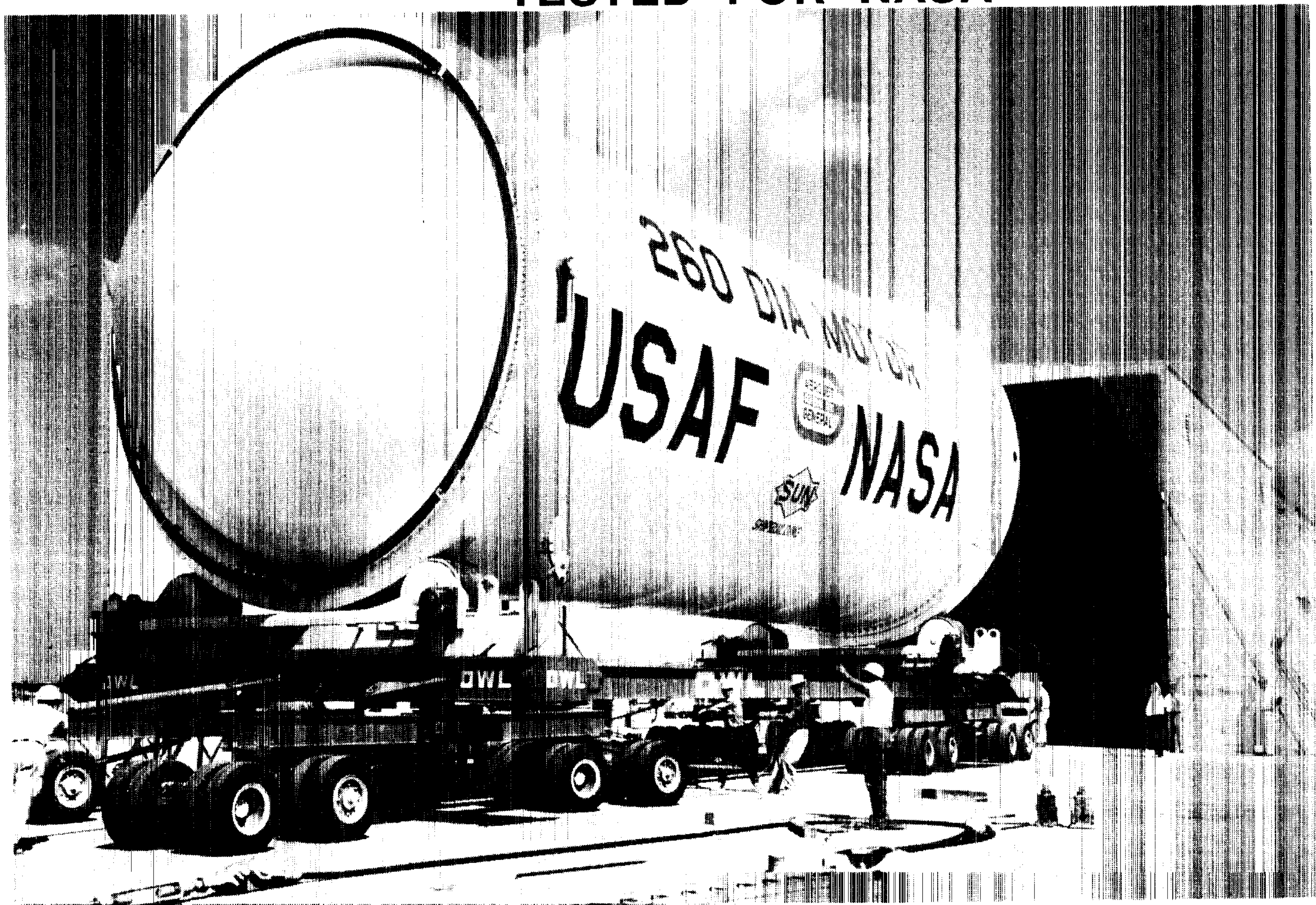
OUR 6.6 METERS (260-INCH) SOLID ROCKET MOTOR IS THE LARGEST EVER BUILT AND WAS SUCCESSFULLY TESTED FOR NASA



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

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SOLIDS EXPERIENCE—LARGE MOTORS

• 6.6 m (260-in.) dia	Contract: NAS3-6284
• Peacekeeper 2.34 m (92-in.) dia	F04704-78-C-0010
• Minuteman	F42600-85-C-0442
• Small ICBM	F04707-87-C-0060
• ASRM (Ongoing Effort)	

SOLIDS EXPERIENCE—CASE MATERIAL AND DESIGN

- Cylindrical
- Tapered
- Spherical
- Oblate Spheroid
- Glass
  - AS-4 Graphite
  - Kevlar-40
  - S-Glass
  - IM-6-Graphite
- Metallic
  - Hi-Strength Steel D6aC, HP9-4
  - 6Al-4V Titanium
  - Aluminum Alloys
- Composite Over Metal

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SOLIDS EXPERIENCE— FUEL RICH PROPELLANTS

- AP/AS/HTPB
- Non-Metallized
  - AN/HTPB/PAMs
- Metallized
  - AN/AP/HTPB
  - AN/B/HTPB
- Scavenger
  - AP/$NaNO_3$/HTPB

AN or Scavenger Required for Low HCl in Exhaust

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HYBRID PROPELLANT STATUS

Propellant Ingredients, Wt %		Baseline Types ANB-3302-4	ANB-3656	Non-Metallized Naphtahlene	PAMS	BAMO PAMS	Metallized Al	Oxygen Rich ANP-3576
AP Oxidizer		61.5	52.5	35.0	43.0	35.0	6.0	82.0
AN Oxidizer		--	--	--	--	--	46.0	--
AS Coolant		16.0	25.0	--	--	--	--	--
Fuel				43.0	35.0	35.0/8.0	20	--
Binder		22.5 (HPTB)	22.5 (HTPB)	22.0 (HTPB)	22.0 (HTPB)	22.0 (HTPB)	28.0 GAP/TMETN	18.0 (Fluoro-carbon)
		100.0	100.0	100.0	100.0	100.0	100.0	100.0
• Experience	1.	32 kg (70-lb) BATES Motors FS Mix Size	0.5 kg (1-lb) Motor Test	—Lab Scale BR ~1.25 cm/sec (0.5 ips)—			2.25 kg (5-lb) Motor Demo	0.5 kg (1-lb) Motor Demo
	2.	Tested With LOX Injector	Demonstrated in ATR				BR 0.76 cm/sec (0.3 ips) at 6.89 MPa (1000 psia)	Demonstrated in Staged Combustion Motor
• HCl, %		13.0	~10	4.5	6.1	4.5	~1.0	18.6
• BR at 6895 kPa (1,000 psi), cm/sec (ips)		1.09 (0.43)	1.09 (0.43)	1.04 (0.41)	1.24 (0.49)	1.30 (0.51)	0.76 (0.30)	1.27 (0.50)

Fuel Rich and Oxygen Rich Compositions Demonstrated With AP Oxidizer, Metallized; Composition With AN Oxidizer Demonstrated With Energetic Binder/Plasticizer. OX Rich System Demonstrated.

SUMMARY

- Solid Propellant Compositions Identified for Hybrid Propulsion
  - Baseline ANB-3302-4
  - Non-Metallized
  - Metallized
  - Scavenger
- Propellant Compositions With AP Successfully Tested
- GAP/TMETN/AN Propellants Successfully Tested
- OX Rich Propellant Successfully Tested

SOLIDS EXPERIENCE— GRAIN DESIGNS

- Finocyl
- Dendrite
- Rod and Shell
- End Burn
- Web in Head
- Combinations of the Above

SOLIDS EXPERIENCE— IGNITERS

- Materials
  - — Propellants
    - Aluminized
    - Reduced Smoke
    - Gas Generator
  - — Pellet Pyrotechnics
    - Black Powder
    - Alelo
    - BPN
  - — Initiator
    - Igniter Cord (Electro Match)
    - Squibs
  - — Location
    - Head End
    - Aft End
    - Surface

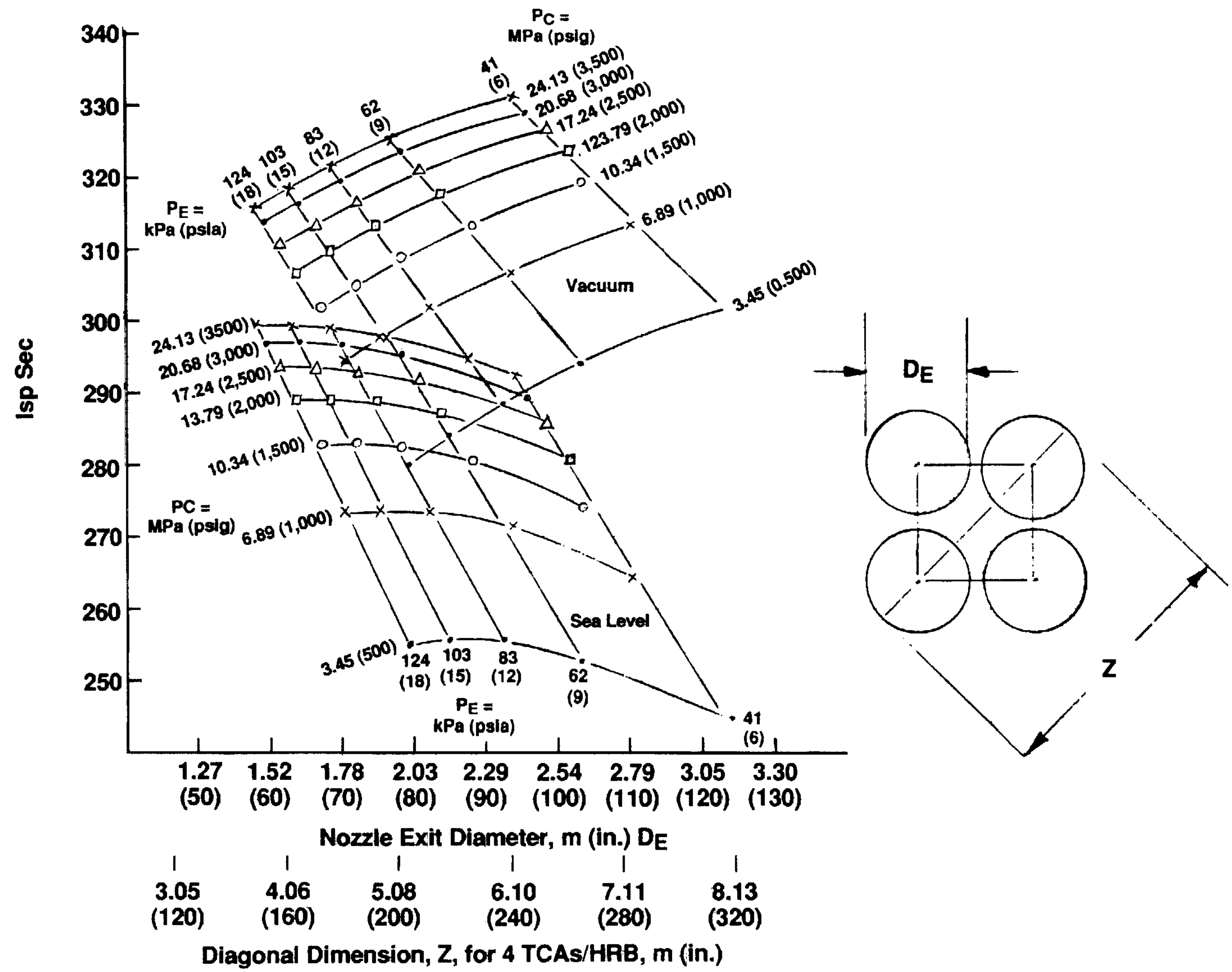
LIQUID PROPULSION EXPERIENCE

- LRB for STS Study (Martin/Aerojet - 1988)
- Aerojet SSME Main Injector
- Aerojet M-1 LO_2 Turbopump and Engine
- Aerojet/NASA OTV Engine With GO_2 Turbopump
- NASA and Aerojet LO_2 Regeneratively Cooled Thrust Chamber Testing

Our LRB for STS Study in 1988 (MMAG Prime Contractor for NASA MSFC) Showed That Large Area Ratio Nozzles Improve STS Payload Performance. It Also Showed That Lower P_c Systems Have Larger Nozzle Diameters When Expanded to the Same Exit Pressure as Higher Pressure Systems of the Same Thrust Level. These Larger Nozzles Will Not Fit the Current Mobile Launch Platform for STS

VACUUM AND SEA LEVEL I_{SP} FOR F_{SL} = 750,000 LBF

T_C = 3478°K (6260°R), M = 24, γ = 1.16



Posts Numbered 2, 4, 6, and 8 Need to Be Moved Outboard to Fit the LRB or HRB

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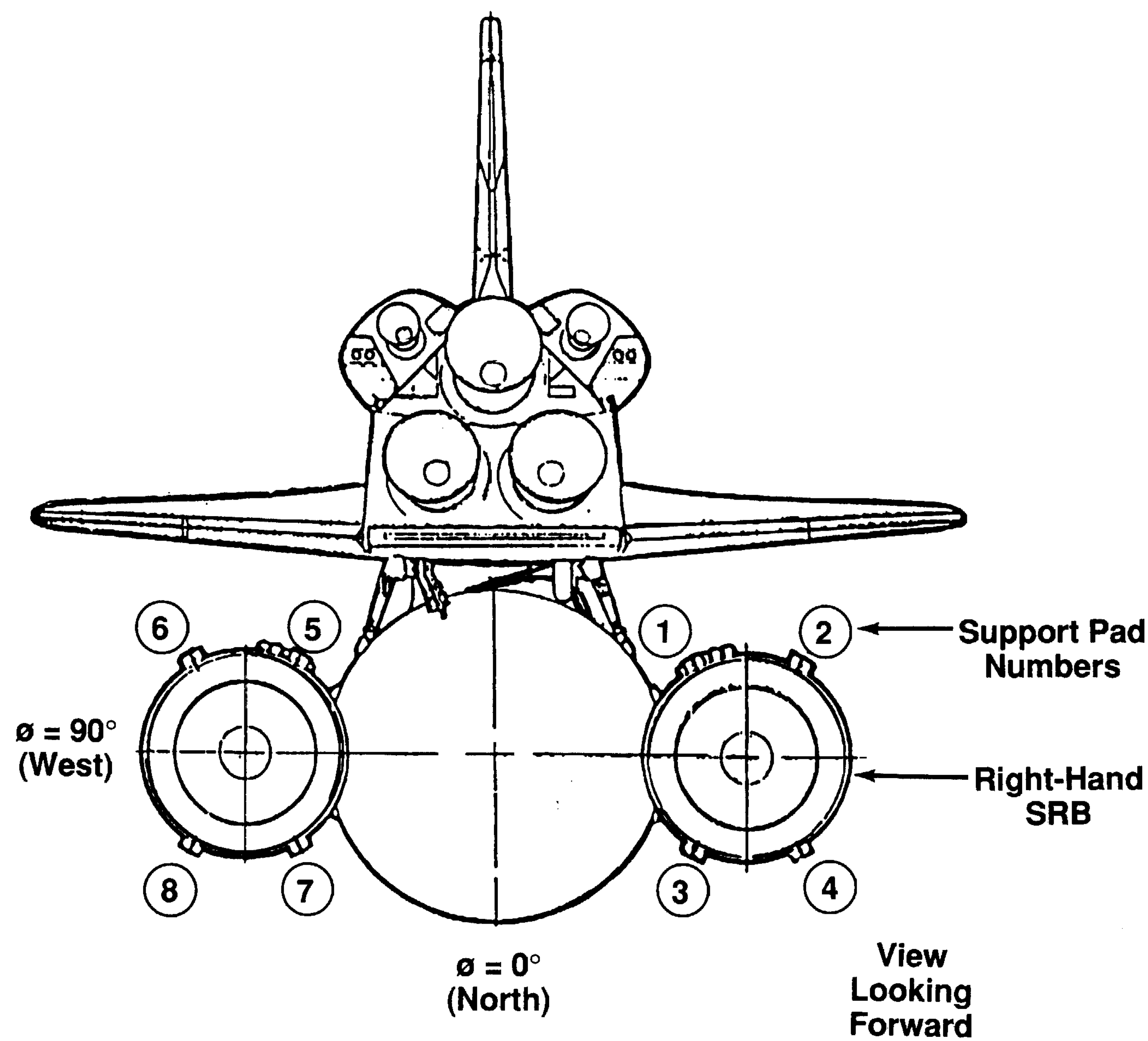
NUMBERING OF FACILITY/SRB SUPPORT PADS



If the Rectangular SRB Support Posts Are Moved Into a Square Pattern Enough Space Is Generated for Four Very Large Diameter Nozzles to Fit Between the Posts. After 30 Years of Use the MLPs May Need Refurbishment, and Modification Costs Would Be Reasonable. **See Page 204**

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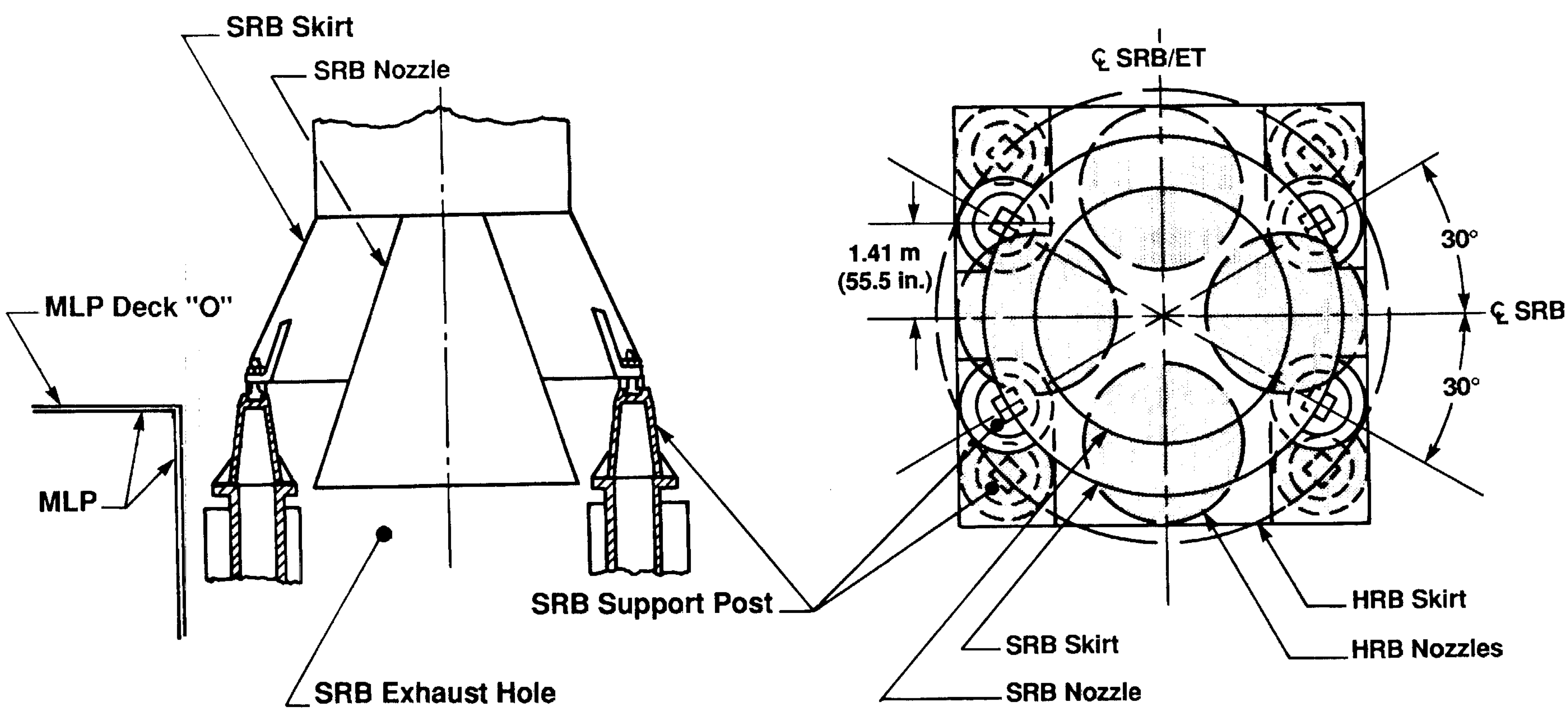
MLP WITH SQUARE SUPPORT POST PATTERN FITS HIGHER PERFORMANCE BOOSTERS



During 1970, Aerojet and Two Competitors Performed 1-Year Contracts for NASA MSFC, Developing Proposals for the SSME. Aerojet Built and Tested Several Engine Components, Including Preburner and Main Injectors. Although the Main Injector Shown Here Was Not Hot Fired, It Was Designed, Built, and Flow/Pressure Tested for Mixing Efficiency and Structural Adequacy. This Design Is Similar to the Successful, Storable Propellant, Vaned, Gas/Liquid Injector Tested at 19.3 MPa (2,800 psi) Pressure During Our ARES (Advanced Rocket Engine - Storable) Contract With the USAF. The SSME Unit Has Nine 40-Deg Pie-Shaped Segments of Platelet Vanes Mounted Within a Liquid Oxygen Distribution Manifold. Gaseous Fuel Flows Between the Vanes and Mixes With Droplets of LO_2 Sprays Formed by Trailing Vane Edge Impinging Element Flow

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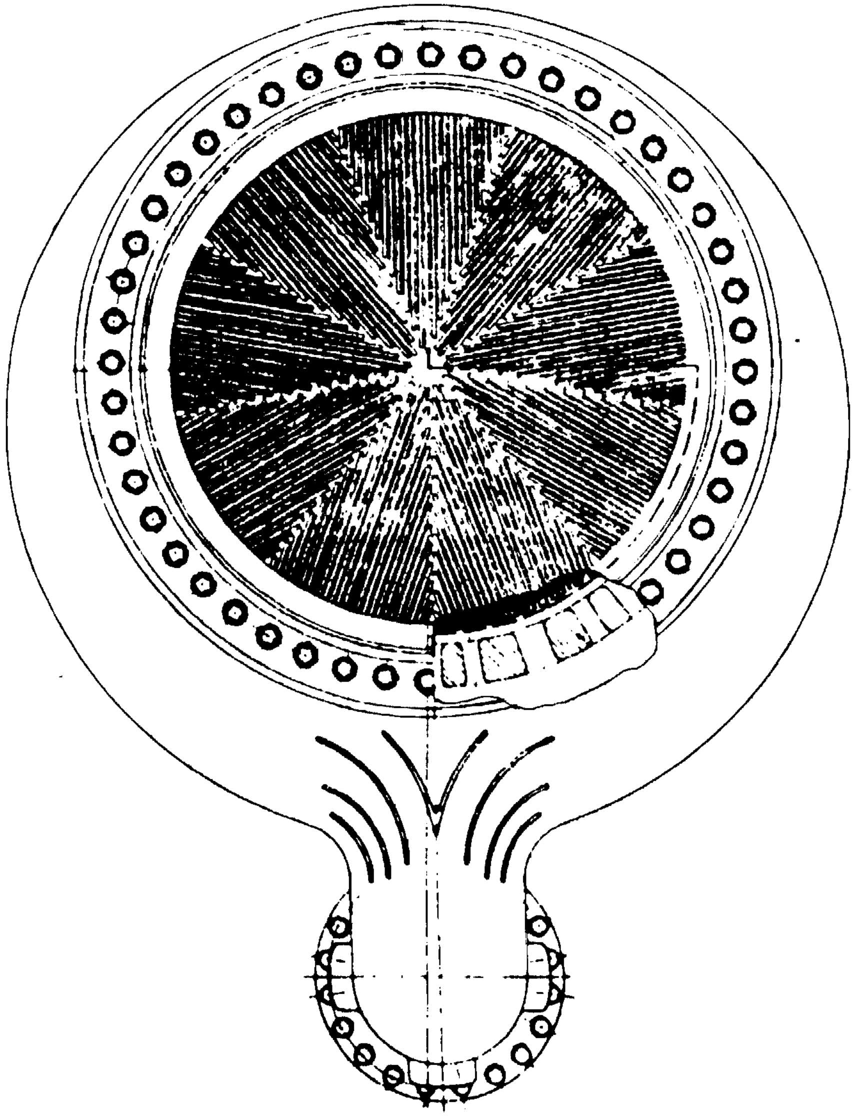
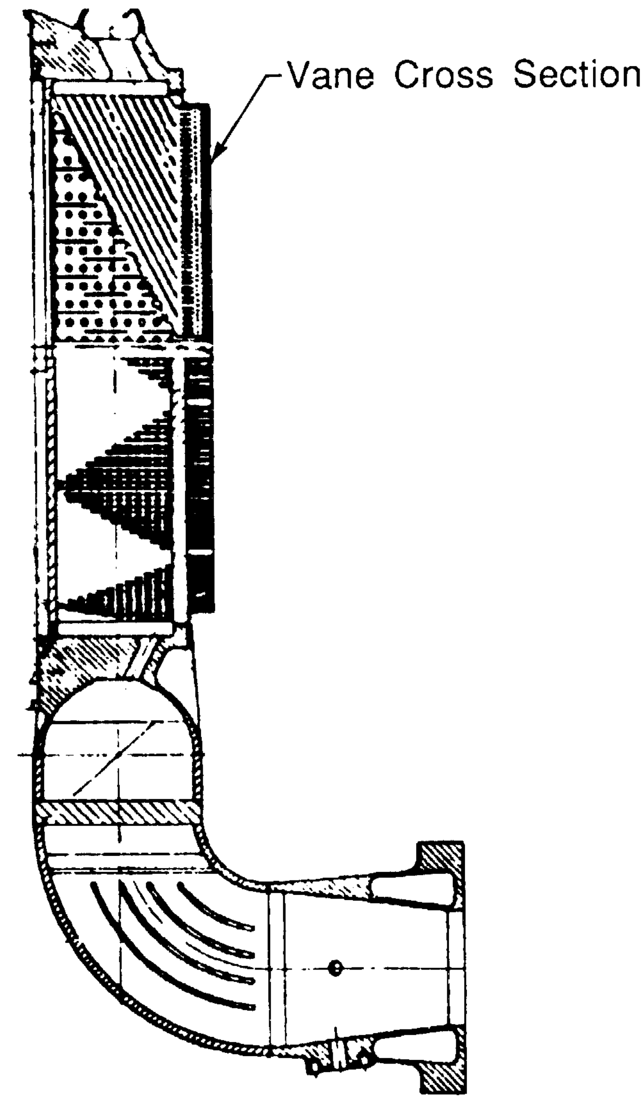
AEROJET SSME MAIN INJECTOR

1.85 MN (415 klbf) Thrust Gas/Liquid Unit



PLAN VIEW



SIDE VIEW AND CUTAWAY

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AIAA PAPER

DESIGN AND TEST OF AN OXYGEN TURBOPUMP FOR A DUAL EXPANDER CYCLE ROCKET ENGINE

P. S. Buckman, N. R. Shimp and F. Viteri
Aerojet TechSystems
Sacramento, California

M. Proctor
NASA Lewis Research Center
Cleveland, Ohio

Abstract

A liquid oxygen (LOX) turbopump with an 860°R gaseous oxygen (GOX) turbine drive was designed for a 3750 lb thrust dual expander cycle rocket engine. This turbopump, which requires no interpropellant seals or system purges, features a 156 hp, single stage, full admission, impulse turbine, an axial flow inducer; a two-stage centrifugal pump with unshrouded impellers; long-life, LOX-lubricated, self-aligning, hydrostatic bearings; and a subcritical rotor design. It is constructed of Monel, a nickel-copper alloy, which has low ignition potential in oxygen. The pump was designed to deliver 34.7 gpm of 4655 psia liquid oxygen at a shaft speed of 75,000 rpm. The dual expander cycle rocket engine and the performance it requires of the LOX turbopump will be discussed as well as the design of the pump, turbine, bearings, and the turbopump rotordynamics. The test program and test results will also be presented.

Table 1 Orbit transfer propulsion basic operating requirements and goals

Requirements	
Propellants - Fuel	Hydrogen
- Oxidizer	Oxygen
Vaccum Thrust (Design Point Range)	10,000 to 25,000 lbf
Engine Mixture Ratio, O/F (Design Point)	6.0
Engine Mixture Ratio, Range, O/F	5.0 to 7.0
Propellant Inlet Temperature	
Hydrogen	37.8°R
Oxygen	162.7°R
Gimbal	±6.0 deg (Square Pattern)
Start Cycle	Chilldown with propulsive dumping of propellants. Engine start with pump inlets at propellant tank vapor pressure.

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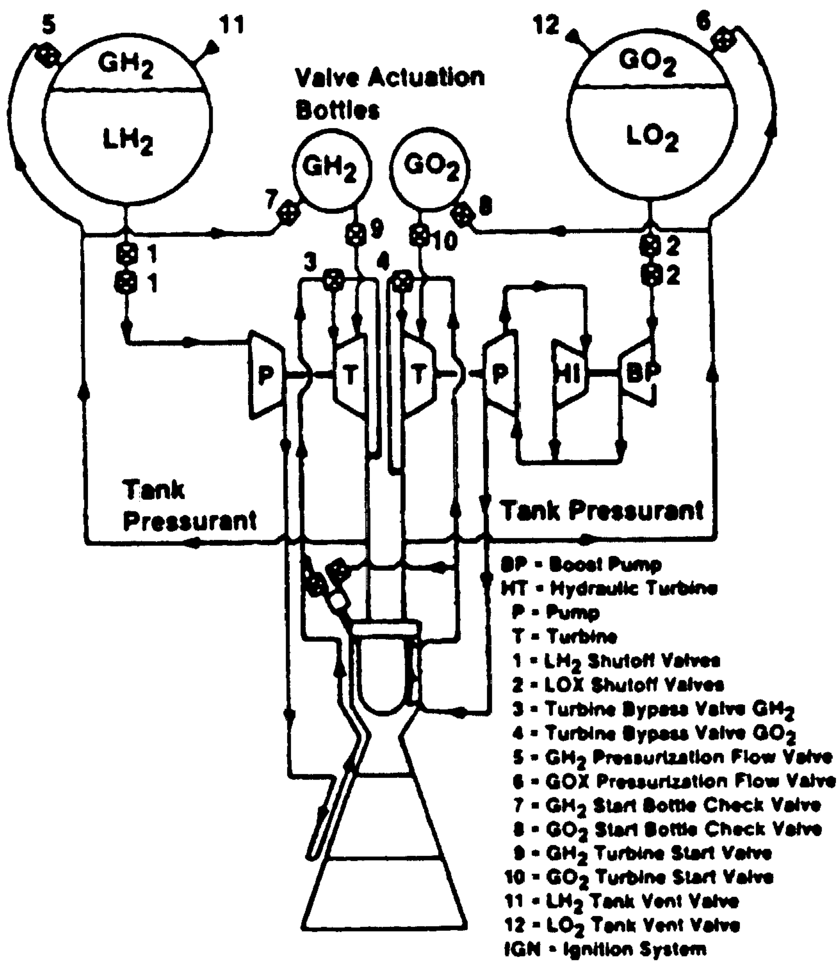
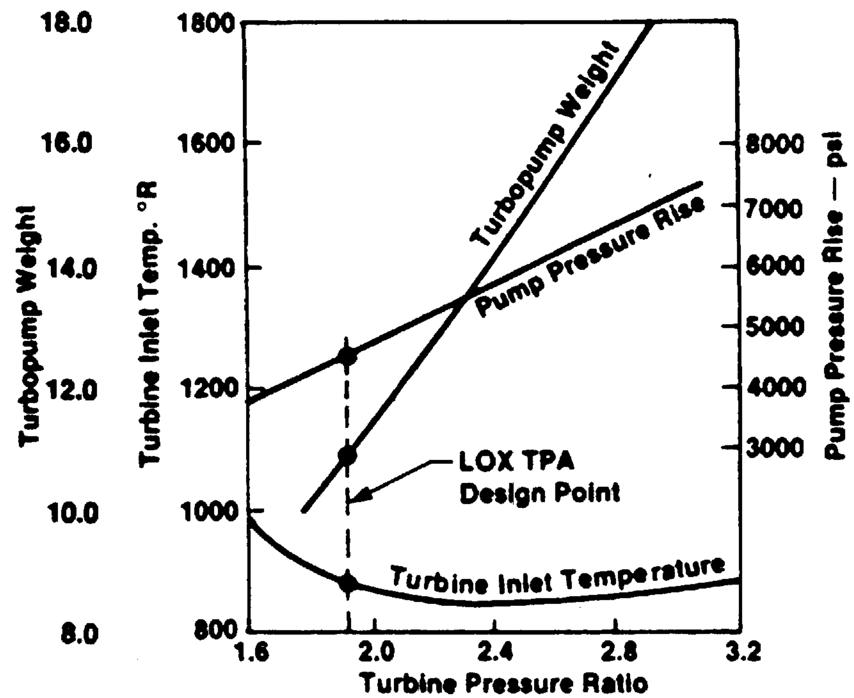
NASA/AEROJET OTV ENGINE USES A LO_2 PUMP DRIVEN BY REGENERATIVELY HEATED GO_2



Flow Schematic of the Dual Expander Cycle Rocket Engine With LO_2 Boost Pump



OTV Oxidizer Turbine Inlet Temperature/ Pressure Rise and Weight Ratio Versus Turbine Pressure Ratio

A pressure ratio of 1.93 was selected as the turbine design point. This value also corresponds to a turbine inlet temperature of 860°R with 7% bypass flow for engine mixture ratio control authority.

Basically the machine performed as predicted with the exception that the maximum test speed attained (approximately 69800 rpm) was limited by available turbine supply pressure. This is illustrated by normalized head-flow curve shown in Fig. 14 for off-design performance in LOX. The maximum pump discharge pressure was approximately 4100 lb/in.^{2} again limited by turbine supply pressure. The bearings performed as intended at steady state speeds supporting high axial loads, and the turbopump operated subcritical throughout test program.

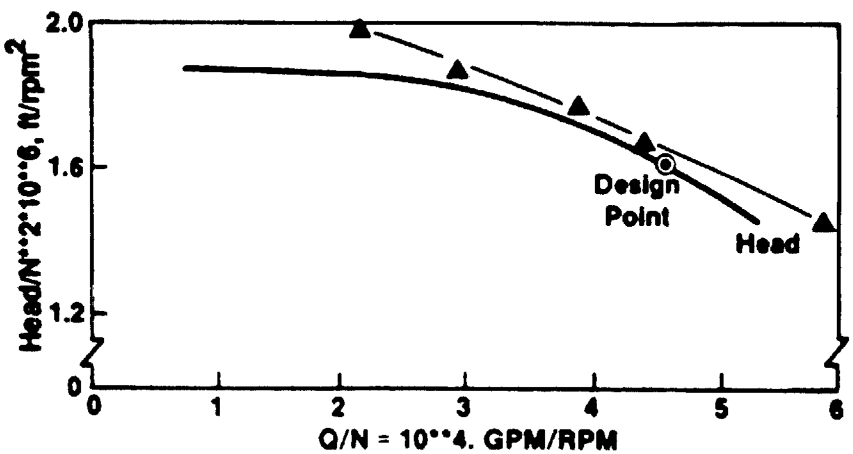


Fig. 14 Predicted OTV performance – Overall performance

A typical test was 60-100 seconds long including start and stop transients. Pressure as a function of time for a typical test is shown in Fig. 15. Unassisted bearing starts with LOX in the pump were demonstrated 14 times and a total run time of approximately 37 minutes was accumulated.

On disassembly, the turbopump components appeared in excellent condition. Some rubbing on the outboard exit land of each journal bearing was evident. The higher loaded first stage thrust bearing only had minor burnish marks. The turbine rotor did not have any marks from rotation verifying concentric operation without contact.

Summary

A liquid oxygen turbopump with 860°R gaseous oxygen turbine drive was designed for a 3750 lb thrust dual expander cycle rocket engine. This turbopump, which requires no inter-propellant seals or purges, features a 156 hp, single stage, full admission, impulse turbine; an axial flow inducer, a two-stage centrifugal pump with unshrouded impellers; long-life, LOX-lubricated, self-aligning, hydrostatic bearings; and a subcritical rotor design. it is constructed of Monel, a nickel-copper alloy, which has low ignition potential in oxygen. The pump was designed to deliver 34.7 gpm of liquid oxygen at a discharge pressure of 4655 psia and a shaft speed of 75,000 rpm. Testing has demonstrated subcritical rotor operation, successful operation of a turbine in ambient temperature gaseous oxygen, unassisted hydrostatic bearing starts, and stable and predictable pump performance over an operating range of 40 percent to 120 percent of design flow to speed ratio (Q/N). Material loss in the bearings was minimal. Although life testing and testing with warm (860R) GOX in the turbine must be conducted, results of testing to date indicate a GOX driven LOX turbopump is feasible.

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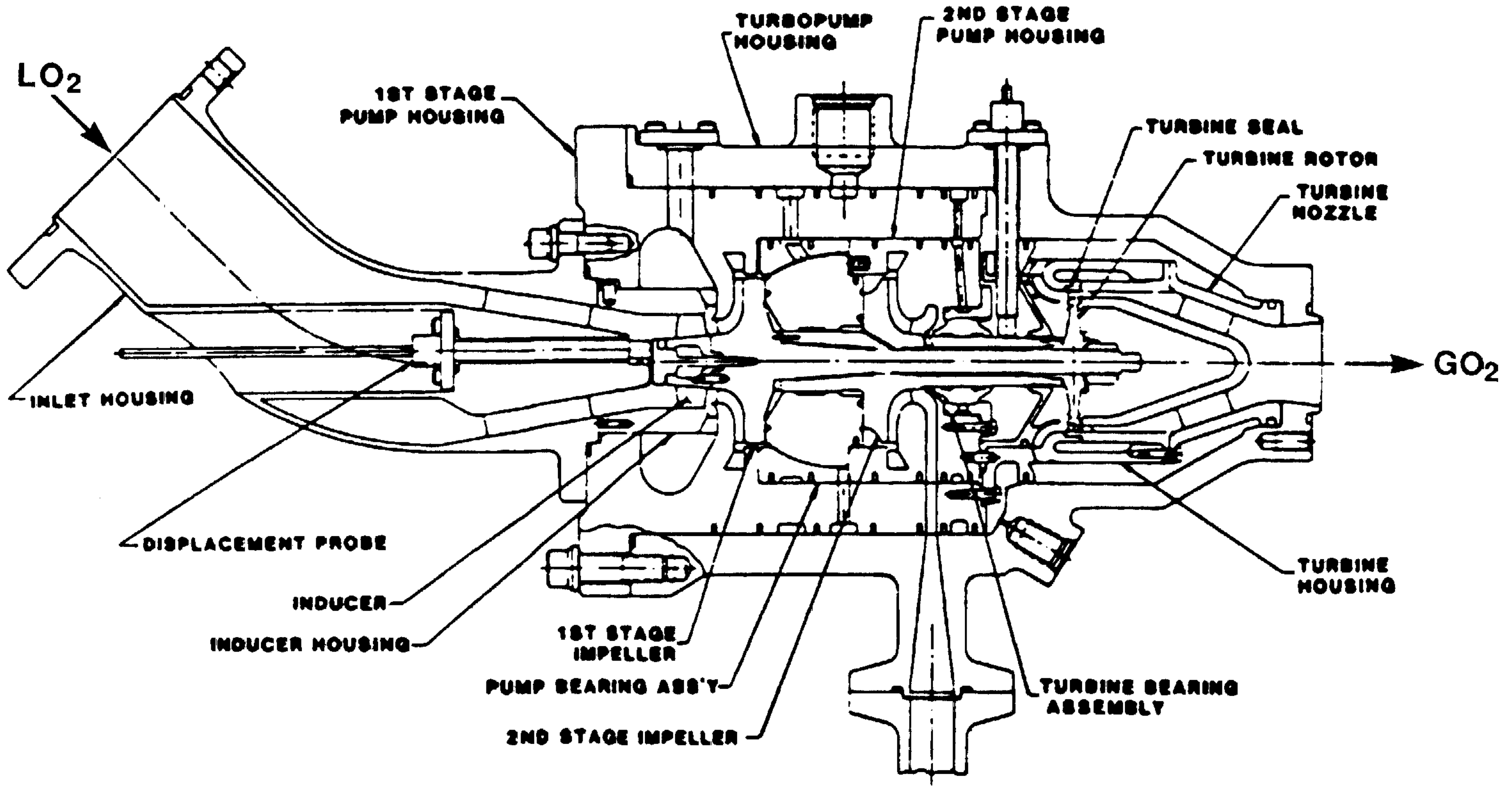
OTV OXYGEN TURBOPUMP ASSEMBLY



Note: Two Stage LO_2 Pump and Single Stage GO_2 Turbine Rubbing Start Hydrostatic O_2 Bearings

ORIGINAL PAGE IS OF POOR QUALITY

The Most Recent NASA Work at Lewis Research Center With LO_2 Regeneratively Cooled Rocket Thrust Chambers Was Reported at the AIAA Conference in Monterey, CA in July 1989. By Elizabeth S. Armstrong, It Is Entitled: "Liquid Oxygen Cooling of Hydrocarbon Fueled Rocket Thrust Chambers." This NASA Summary, Shown Below in the Following Two Pages, Was the Best One Available Prior to the Monterey Conference

NASA HAS DEMONSTRATED LO_2 COOLED THRUST CHAMBERS

Summary of Results

Five Thrust Chambers With Identical Coolant Passage Geometries Were Tested With LOX/RP-1 as the Propellants and LOX as the Coolant. Three of These Thrust Chambers Were Tested at 4.14 MN/m^2 (600 psia) Chamber Pressure and Over a Mixture Ratio Range of 2.25 to 2.92. One Thrust Chamber Was Tested at 8.274 MN/m^2 (1200 psia) Chamber Pressure Over a Mixture Range of 1.93 to 2.98. Two of the Thrust Chambers Were Tested at 13.79 MN/m^2 (2000 psia) Chamber Pressure Over a Mixture Ratio Range of 1.79 to 2.68. The Results of These Tests Were as Follows:

1. Successful Cooling With LOX Was Demonstrated

2. One Chamber Was Cyclically Tested 93 Times. During This Testing, Cracks Appeared in the Hot-Gas Wall That Permitted Oxygen to Flow Into the Combustion Region With No Catastrophic Failures. With This Chamber, More Than 22 Cyclic Tests Were Made After the First Through-Crack Was Observed With No Apparent Metal Ignition or Distress

3. The LOX Passing Through the Crack in the Hot-Gas Wall Did Not React With the Carbon Layer at the Throat on the Combustion Wall, Thereby Raising the Metal Wall Temperature to Its Ignition Temperature and Causing a Catastrophic Failure. It Also Did Not React Directly With the Metal Wall

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NASA HAS DEMONSTRATED LO_2 COOLED THRUST CHAMBERS

Summary of Results (Cont.)

4. The Thrust Chamber Wall Cracks That Formed as a Result of the Cyclic Testing With LOX as the Coolant, Appeared to Have Similar Characteristics as Those From a Previous Program Where Liquid Hydrogen Was the Coolant

5. The LOX Cooling of LOX/RP-1 Propellants Was Very Similar to the LOX Cooling of Hydrogen/Oxygen Propellants

6. At a Nominal O/F of 2.8, Soot Thickness Decreases as Chamber Pressure Increases, Except in the Cylindrical Portion of the Thruster Where it Remained a Constant Thickness

7. Soot Deposition as the Least in the Throat Region at All Chamber Pressures and Mixture Ratios

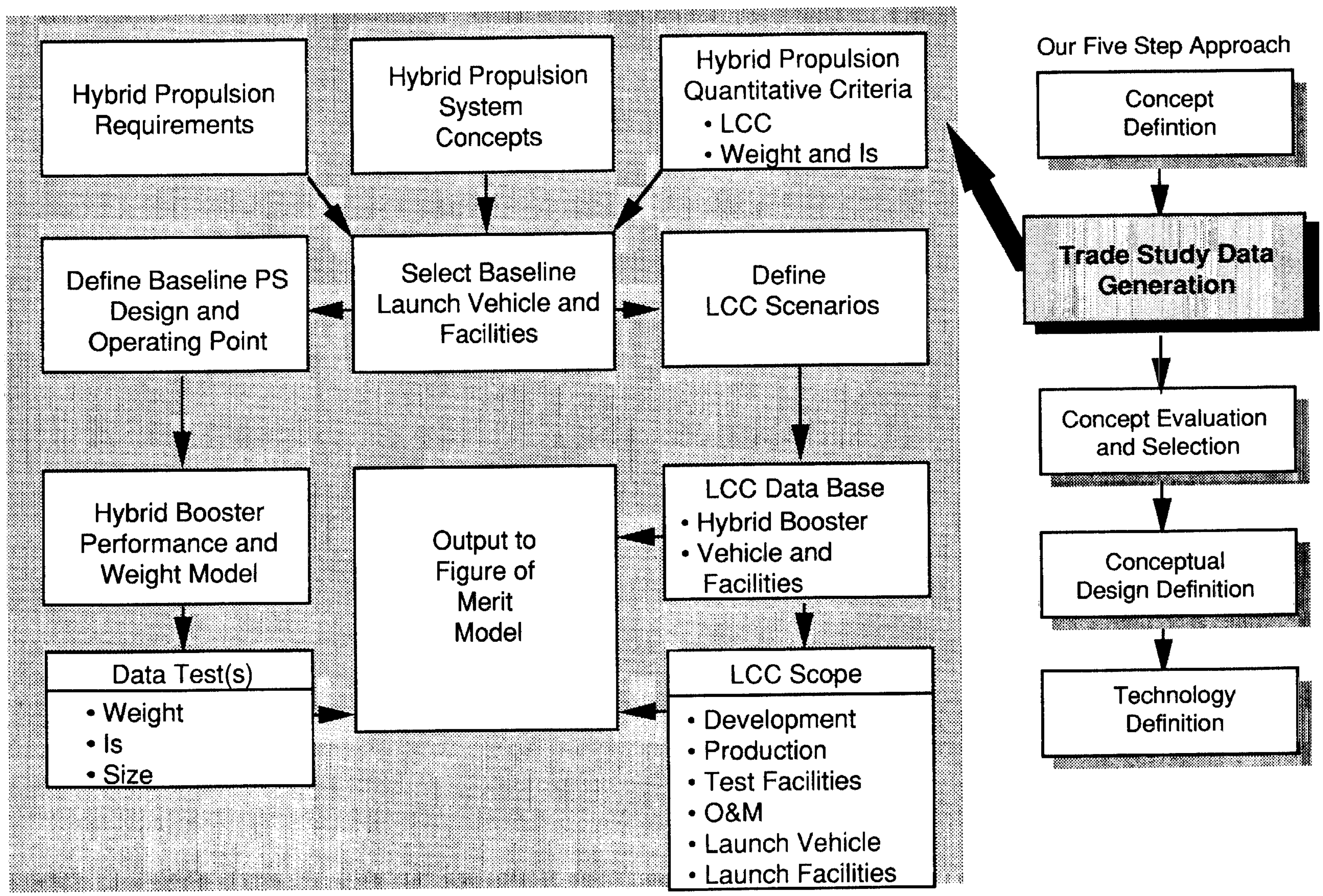
8. Soot Thickness Decreased at a Given Thrust Chamber Axial Location as Mixture Ratio Increased in the Range From 2 to 3

In Task 2, Trade Study Data Generation, We:

- Selected Baseline Launch Vehicle and Facilities (STS)
- Defined Our Baseline HRB Propulsion System Design and Operating Point Against Which to Rate All Concept Changes
- Developed an STS/HRB Size, Weight, and Performance Computer Model
- Calculated STS and HRB Weight and Size, as Well as I_{sp} and Payload to LEO for Each Concept to Be Evaluated
- Defined Life Cycle Cost Scope, HRB Scenario Data, Our Cost Data Base, and Costing Methodology

We Begin This Section With Weight Data Results, and Work Backward Toward Our Baseline Selections, Followed by Our LCC Preparation Work

Our Trade Study Data Generation Process is Computerized, Responsive to the SOW Ranking Criteria, and Supportive of Our Concept Ranking Methodology



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HRB PERFORMANCE AND WEIGHT MODEL

- Level 1 — Propellants
- Level 2 — Combustion
- Level 3 — Subsystems

LEVEL 1 WEIGHT DATA-PROPELLANTS

This Is an Example of Our Weight Result Chart. It Shows How STS Payload Weight to LEO and Gross Lift-Off Weight (GLOW) Are Affected by the Propellant Combinations Burned by Our Baselined Turbopump Fed Large HRBs. Upper Left Hand Candidates Are Favored. Solid Propellants 8B and 8C Are Similar to Propellant 8 (All Are Burned With LO_2), Except That They Have No Solid Oxidizer in Them (and 8C Contains Aluminum). All HRBs Deliver the Same (Required) Total Impulse and Thrust

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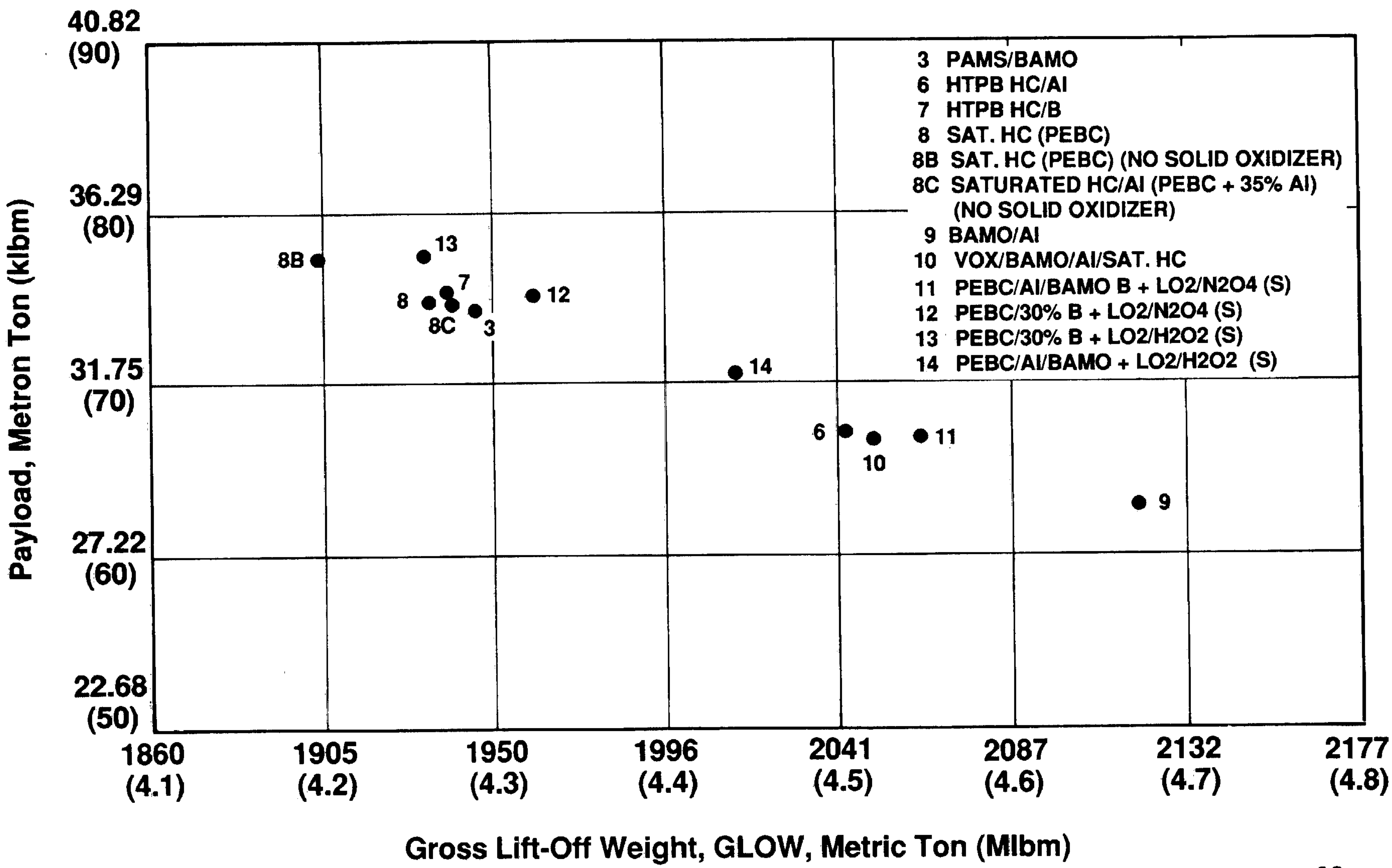
TURBOPUMP FED (TF) HRB, EFFECT ON STS PAYLOAD AND GLOW



This Figure Is Similar to the Previous One, but It Characterizes Weight Differences With Pressure Fed HRBs. Note That the Relative Positions of All Propellant Combinations Are the Same on the Two Charts and That the Pressure Fed HRB Payloads Are Less Than the Turbopump Fed Counterparts

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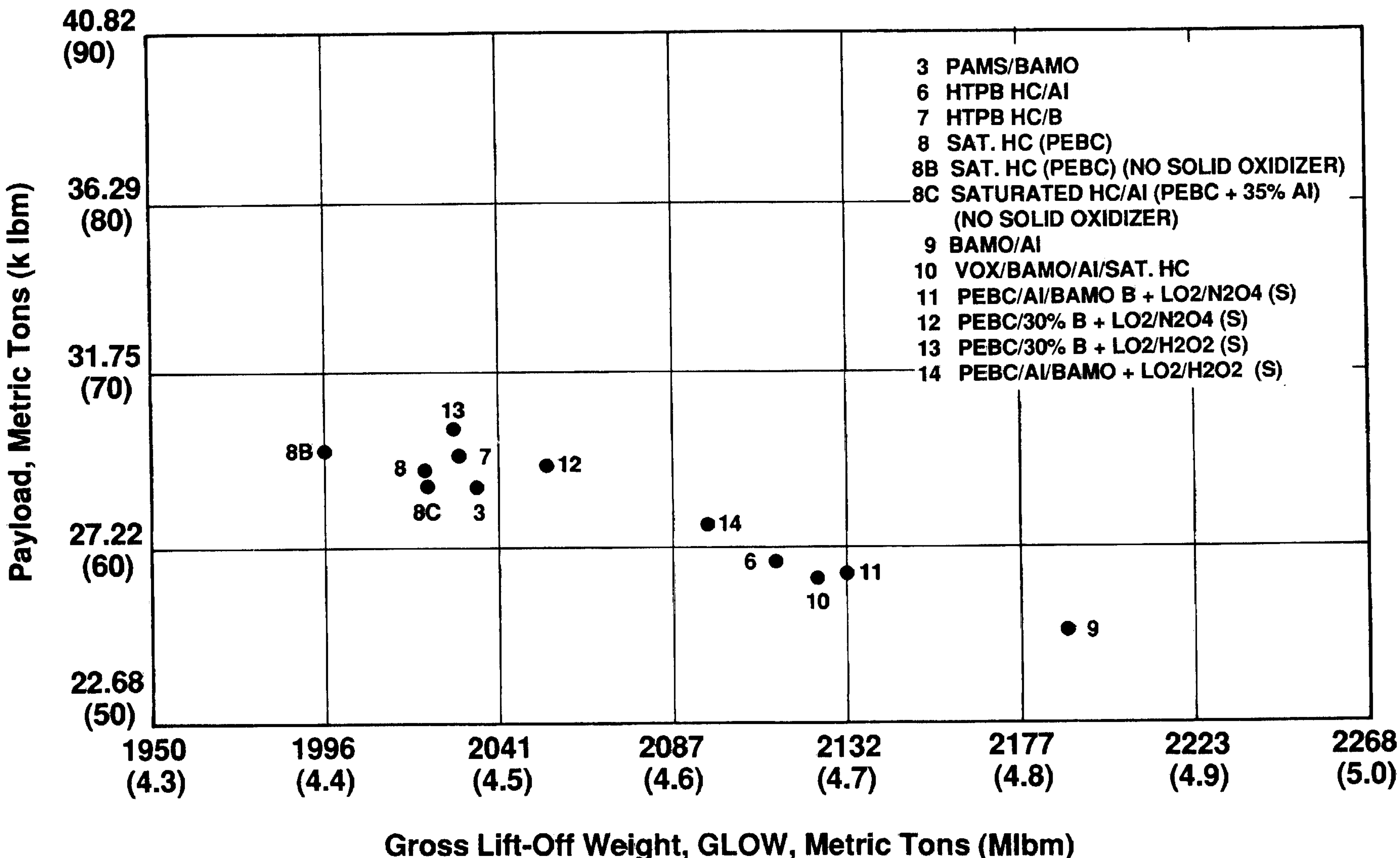
PRESSURE FED (PF) HRB, EFFECT ON STS PAYLOAD AND GLOW



LEVEL 2 WEIGHT DATA- COMBUSTORS

This Weight Results Chart Shows the Impact of Combustor Configuration Selection on Our Baselined STS Payload to LEO and GLOW With Turbopump Fed Large HRBs. The Chart on Page 46 Shows the Six Basic Configurations Considered, of Which Only A, B, D, and E Passed the Screening. All HRBs Burned Propellant 8 or 8b, as Appropriate, With LO_2, and They All Used the Same Bleed Cycle Turbine Drive and Delivered the Same Thrust and Total Impulse.

Concepts With Aft LO_2 Injectors (D and E) Showed Considerably Greater Payload and Lower GLOW, Because the Combustion Efficiency (and I_{sp}) of the Others Were Nearly 10 and 20% Less. This Is Caused by Fuel/LO_2 Combustion in Large Volumes With Poorly Controlled Mixing as Is Shown in the Following Charts

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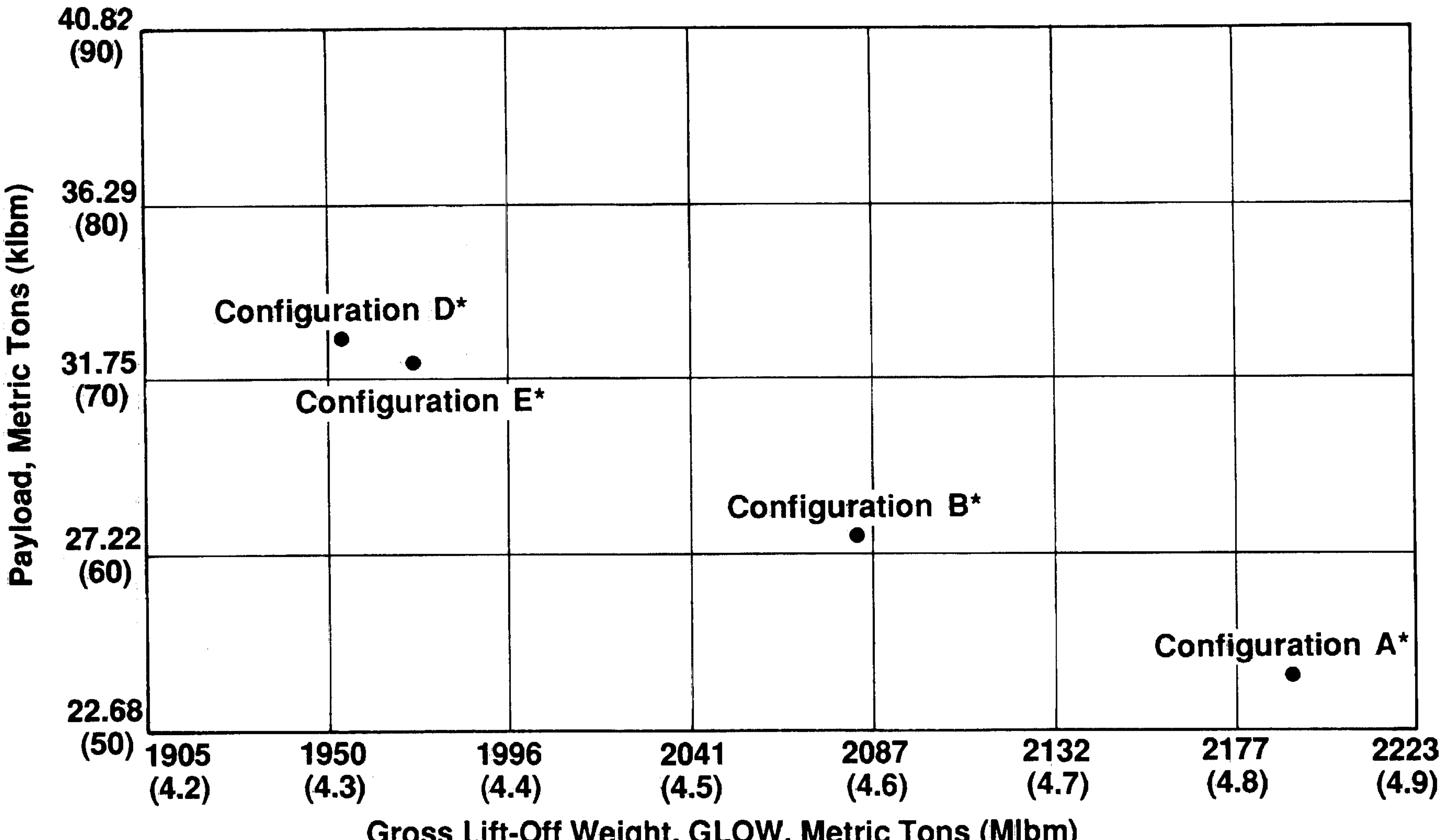
LEVEL 2 TURBOPUMP FED (TF) HRB SUBSTUDIES



*Refer to Chart on Page 46, "Hybrid Concepts"

We Performed a Propellant Mixing Analysis to Determine the Oxidizer/Fuel Mixture Ratio Throughout the Combustor Volume of a Large HRB Grain Port at an Instant of Time. GO_2 Is Injected at the Front End of the Port and Fuel Vapor Evolves From the Solid Grain Surface

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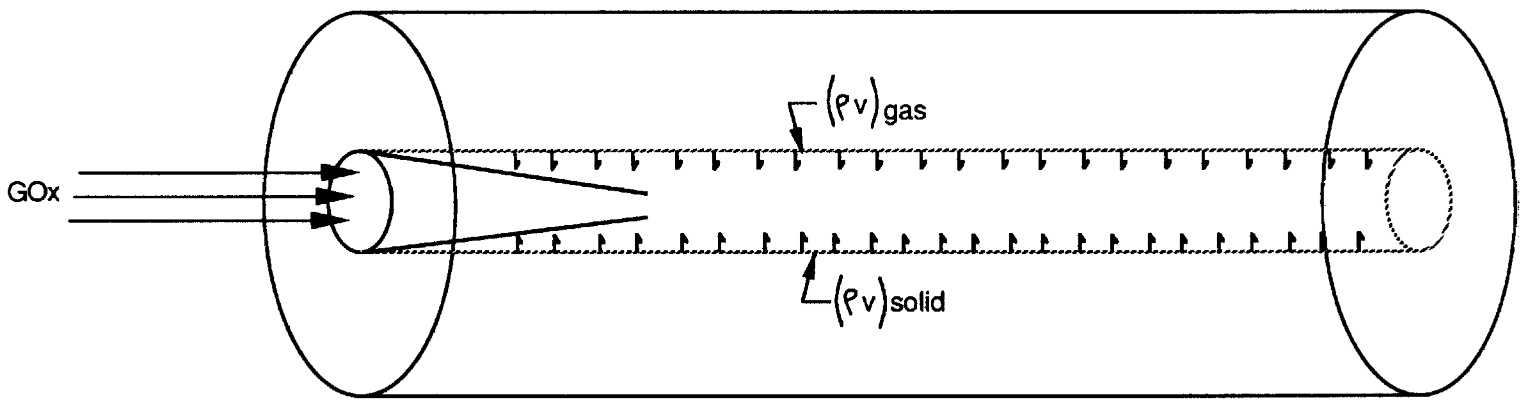
CONVENTIONAL HYBRID SIMULATION



$$V_{injection} = \left(\frac{\rho_s}{\rho_g}\right) V_{solid} \qquad V_{solid} = CP^n$$

3.15.0.19

Mixing Analysis Results Show That the Mixture Ratio Varies Very Greatly Throughout the Combustion Volume

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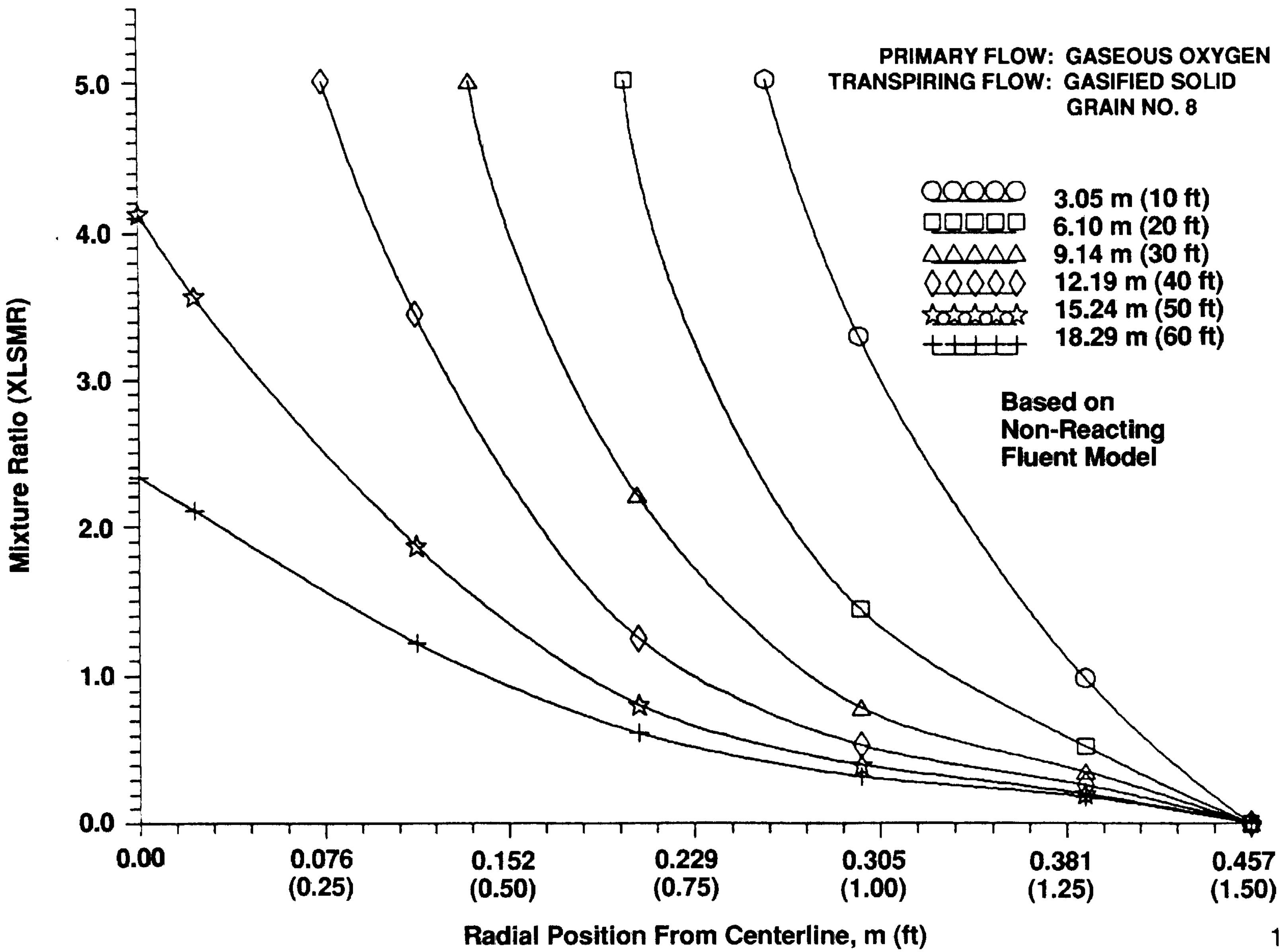
PROPELLANT MIXING ALONG A HYBRID ROCKET CORE



This Is the I_{sp} Versus MR Plot for Our Propellants (No. 8 + LO_2)

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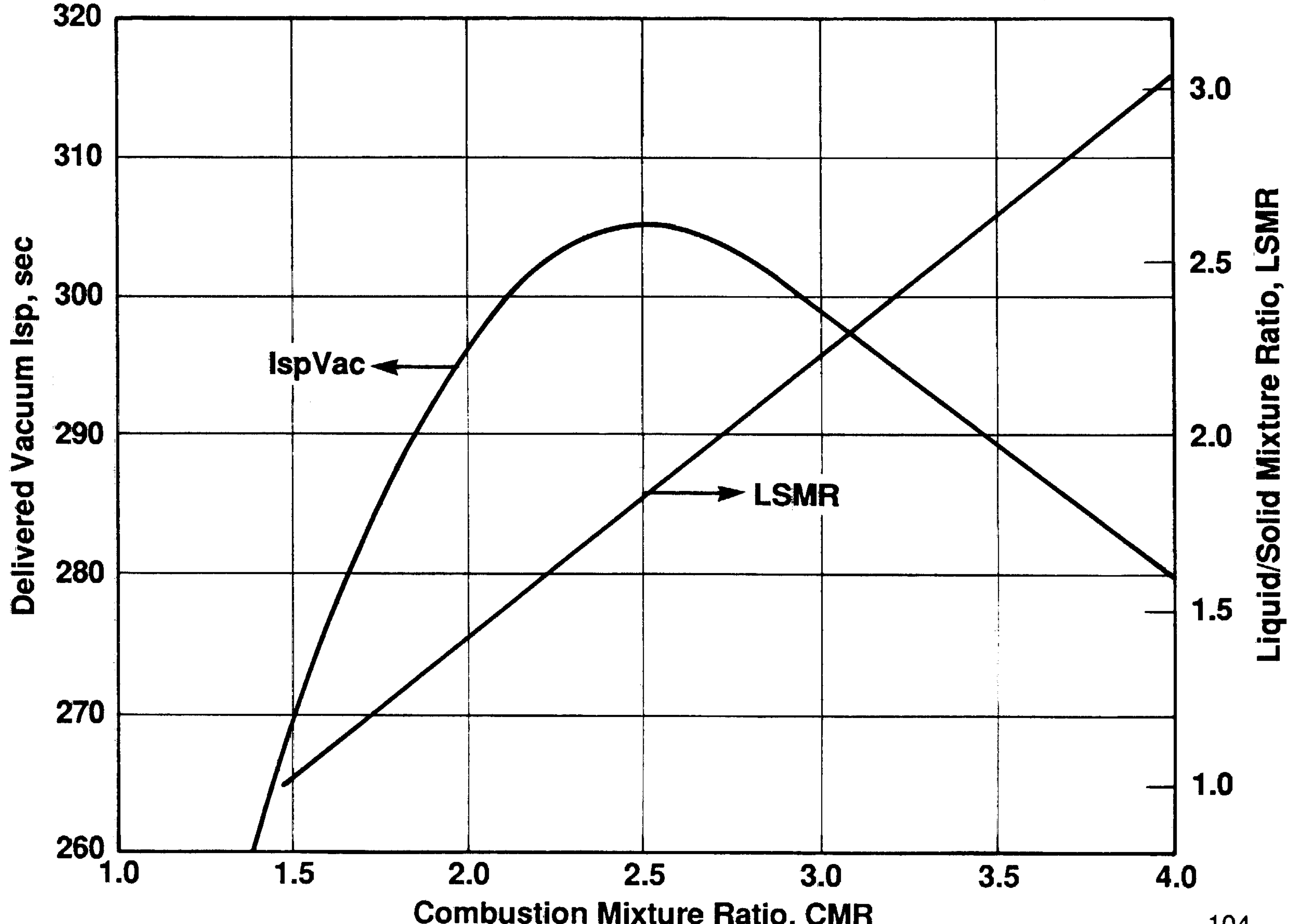
TURBOPUMP FED (TF) HRB FOR STS

Grain #8 Sat. HC (PEBC) + LO2
Pc = 11.72 MPa (1,700 psia)/Pe = 41.37 kPa (6.0 psia)



Notice That Wide Mixture Ratio Variations Cause Low Average Specific Impulse. These Are Mixture Ratio Distribution Losses (MRD). In Our Study, the MRD Losses Turned Out to Be About 50%; However, the Model Did Not Include Combustion Tolerance. Our Algorithm to Approximate Combustion Effects Predicts about 25 to 30% Losses With Combustion. To Make Conservative Predictions, However, We Used 20% Losses for the Conventional Hybrid (Configuration A of Pages 46 and 101) and Only 10% Losses for the Quasi-Hybrid Concept B (Which Contains Some Oxygen in the Solid Propellant Grain). Conversely, We Used 1 to 3% MRD Losses in the Configuration D and E Concepts

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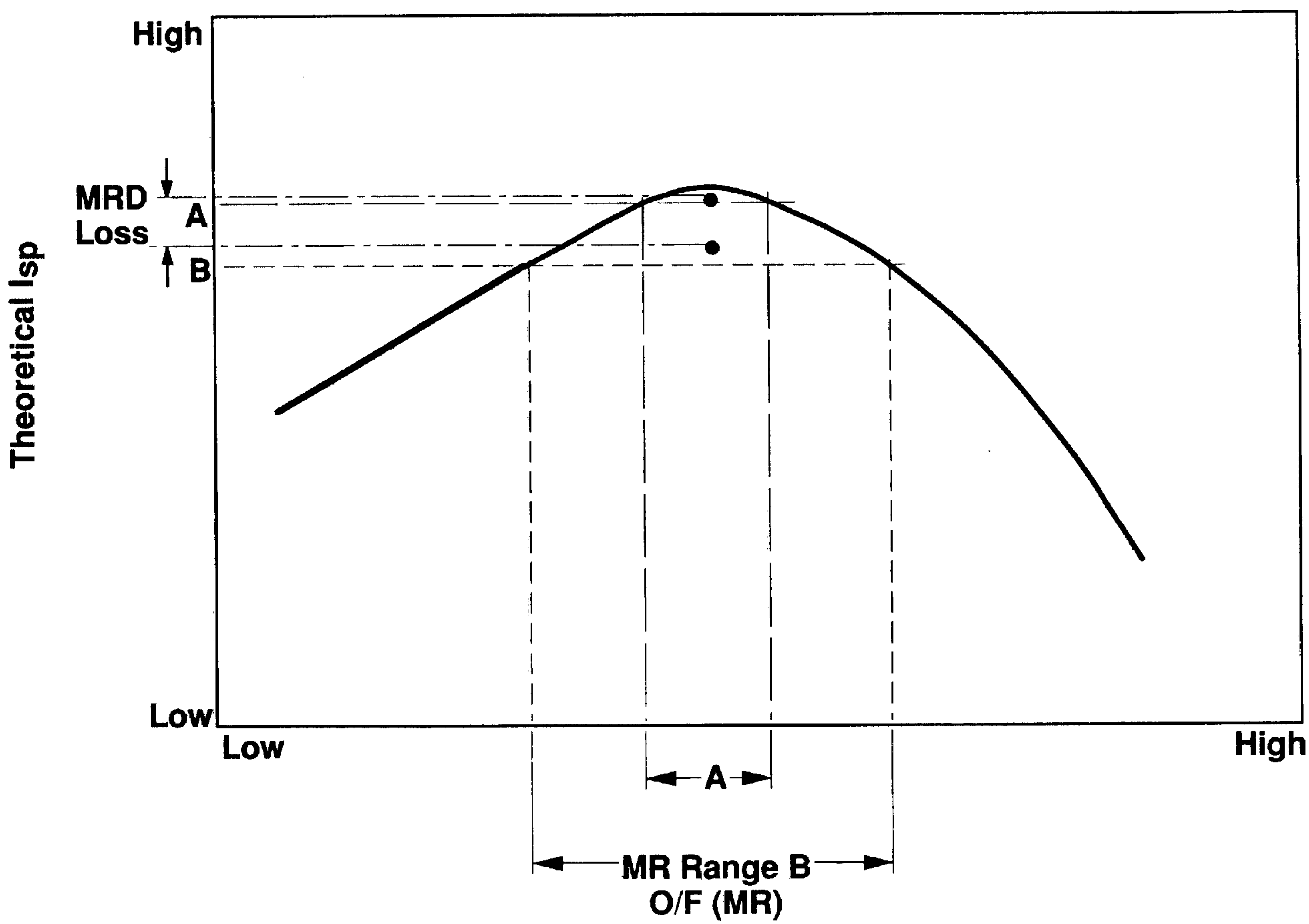
WIDE MIXTURE RATIO VARIATIONS YIELD LOW SPECIFIC IMPULSE BECAUSE OF MIXTURE RATIO DISTRIBUTION LOSSES



M13 HD-038

LEVEL 3 WEIGHT DATA-SUBSYSTEMS

This Chart Is an Example of Our Level 3 Weight Data. It Shows That Bleed Cycle Turbopump Fed HRBs Provide More Payload to LEO at Lower GLOW Than Do Topping Cycle Pump Fed or Pressure Fed HRBs. When Compared With the Next Chart, Which Describes the Generic Weight Results of Our Model Versus I_{sp} and HRB Mass Fraction, We See That the Pressure Fed Weights Suffer From Poor Mass Fraction, and the Topping Cycle From Poor I_{sp}.

The Topping Cycle Should Have Better I_{sp}, Because It Has No Turbine Drive Losses, but Its Gas-Gas Injector Has a Poorer Mixing Efficiency Than Does the Bleed Cycle Counterparts (Because We Are Not Burning H_2, but HC). The MRD Losses This Generates More Than Outweigh the Turbine Drive Losses of the Bleed Cycle.

We Produced Weight Data of This Type for Every Concept Selection Study as Input for the LCC-Based Scoring Model

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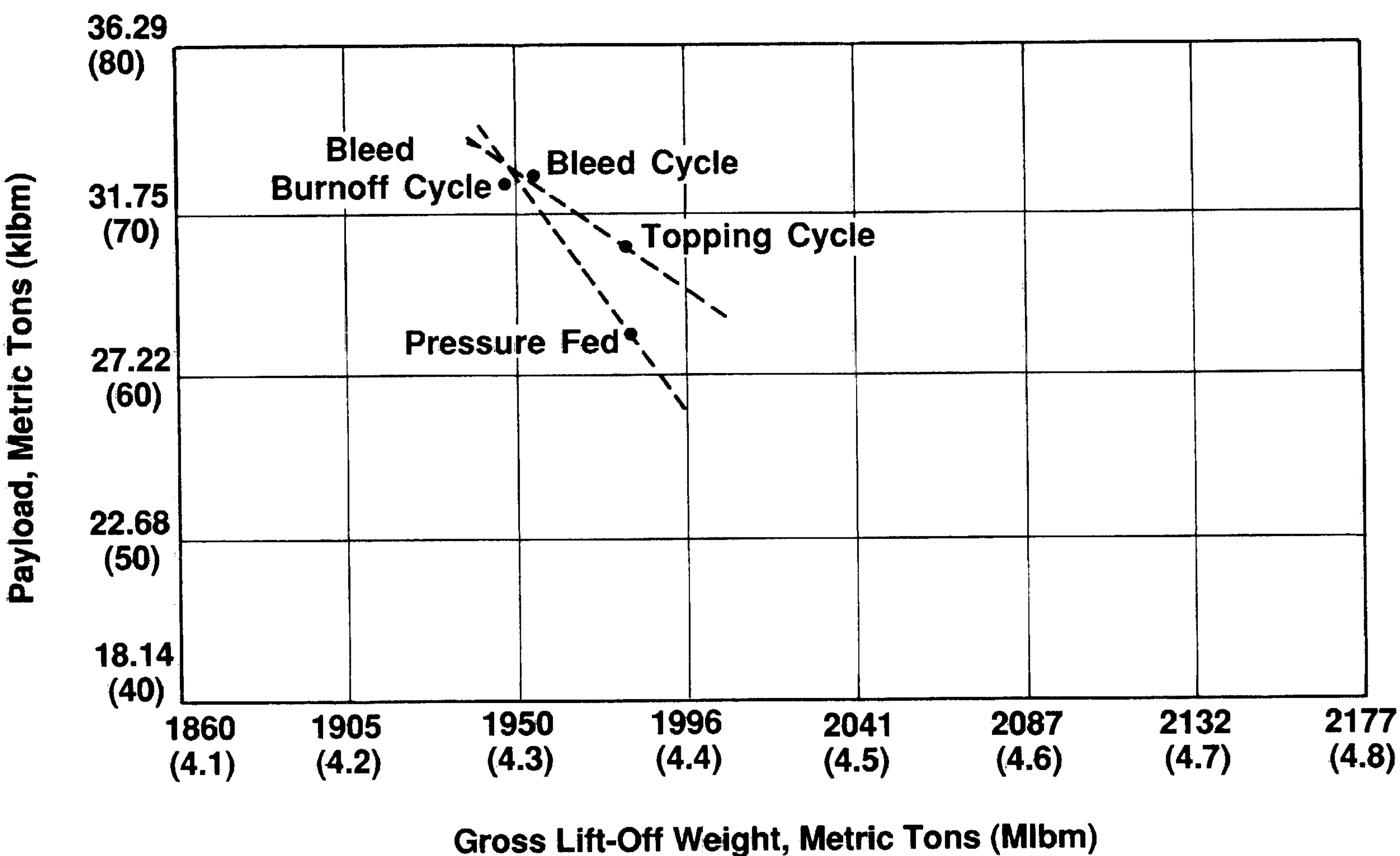
LEVEL 3a PRESSURE FED (PF) AND TURBOPUMP FED (TF) HRB SUBSTUDIES



This Chart Displays the Generic Capability of Our Weight Model. It Is Misleading Only in That Our Model Does Not Consider Mass Fractions, but Rather Weights Themselves. The HRB Propellant Mass Fraction Parameter Is Used for Graphic Convenience Only (MF = Propellant Weight/HRB Wet Weight)

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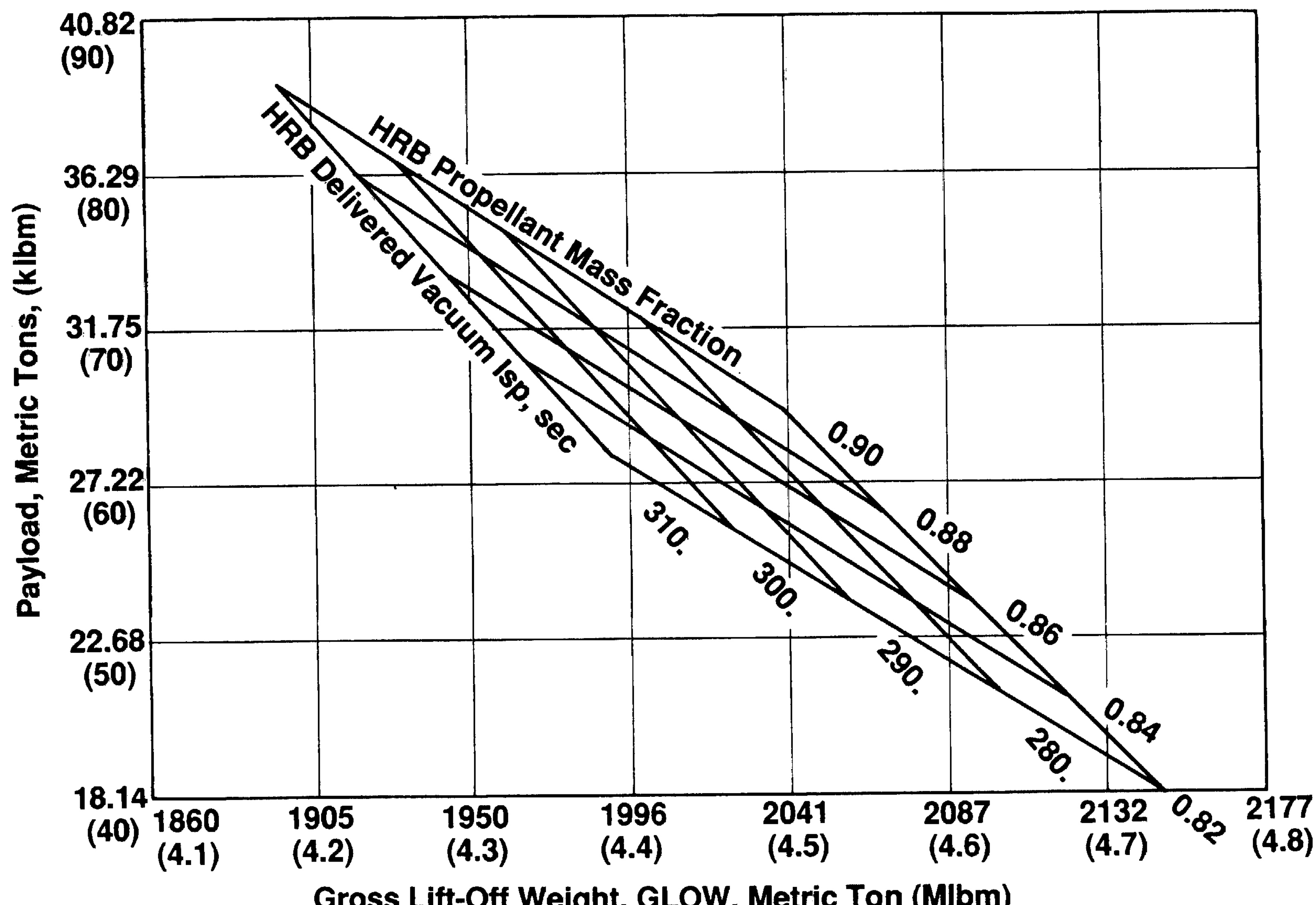
HYBRID ROCKET BOOSTER (HRB) PARAMETERS

Effect on STS Payload and GLOW



HRB PERFORMANCE AND WEIGHT MODEL

We Created a Computer Program That Modeled Our HRB and the STS. It Used a Liftoff Thrust/Weight-Corrected, Ideal Burn Velocity Model to "Fly" the Payload to LEO. This Approximation of a Complete Flight Simulation Gives Good Relative Results Case to Case That Was Needed for Selections With Similar Flight Trajectories. GLOW and Payload Were the Principal Variables Other Than the G Losses That We Corrected for.

Weight Data Generated Was the Design Data Source for Subsequent Cost Data Generation, and Size Output Guides HRB Design.

Input for the Code Was Derived From Several Sources, Including: NASA Requirements, Baselined Vehicle Weights and SSME Performance, Propellant Properties and Performance Data From Our SPP Computer Code, and Standard Ideal Velocity to LEO Data

GENCORP AEROJET

HYBRID BOOSTER (STS) PERFORMANCE AND WEIGHT MODEL

Output:

- STS Payload Weight to LEO
- STS Gross Liftoff Weight (GLOW)
- HRB Liftoff Weight (BLOW)
- Sea Level Thrust and Liftoff Thrust/Weight
- Vehicle Level Weight Breakdown
- HRB Length

Input:

- HRB Total Impulse and Vacuum Thrust
- STS Weights, Thrust, and Specific Impulse
- HRB Specific Impulse Data and Propellant Density (SPP)
- HRB Weight Data
- Ideal ΔV to LEO

SPP Is a Standard Computer Code at Aerojet Used by Theoretical Solid Propellant Chemists

SPP THERMODYNAMIC CODE

Provides:

- Theoretical Combustion Temperatures
  - Specific Impulses
  - Combustion Products
  - Molecular Weights
  - Densities

Applicable to:

- Fuel Rich or Oxidizer Rich Solid Grains
- Hybrid Performance With Liquid Oxidizing Agents or Fuels

CONCLUSIONS FROM SPP DATA

- Saturated Hydrocarbon: PEBC/Tufflo With LOX = Best Hybrid Theoretical I_{sp} Performance, but Lowest Densities
- PAMS/Naphthalene/Tufflo May Yield Lower Combustion Temperatures and Less or No Char in the Primary Grain (Demonstrated With AP)
- Aluminum and Boron Yield High Primary Grain Temperatures and Condensed Phases in the Primary Exhaust
- Scavenger Type Oxidizer Yields NaCl or LiCl Condensed Phases in the Primary Exhaust (Rather Than HCl)
- Calculations With Densified LOX Show Decreased Performance but Increased Densities With Aluminum Fuel Rich Grains
- Performance of Oxidizer-Rich Grains Yields Low I_{sps}

The Next Three Pages Show the Output of Our Weight Model. The First Page Contains Input Data and HRB Specific Impulse and Propellant Weights. It Also Shows the Case Identification Data; This One Is the Selected Configuration

GENCORP AEROJET

TURBOPUMP FED (TF) HRB FOR STS PRELIMINARY DESIGN

TURBOPUMP FED [TF] HRB, 8-9-89,
CASE: 8 D EBBC - 1,4 TCAs, Pe = 41.37 kPa (6.0 psia),
PROPELLANT: GRAIN #8 SAT. HC [PEBC] + LO2

TOTAL IMPULSE, N. SEC (LBF.SEC)	FT	=	1.4234309E+09 (0.320000E+09)
NUMBER OF TCAs PER HRB	TCAS	=	4.00000
HRB ENGINE OUT FACTOR	Z	=	1.00000
ROCKET ENGINE THRUST/WGT RATIO	HRBETW	=	171.110
TANKAGE COEFFICIENT	CTKG	=	367.109
HRB TANKAGE DENSITY, kg/m^3 (LBM/CF)	RTKG	=	81.60 (5.09385)
GAS INJECTOR DELTAP, kPa(d) (PSID)	DPJGAS	=	1172 (170.000)
GRAIN CASE PRESSURE, MPa (PSIA)	PTKG	=	12.89 (1,870.00)
LO2 INJECTOR DELTAP, kPa (PSID)	DPJLO2	=	2344 (340.000)
LO2 TURBOPUMP DEL. PRESSURE, MPa (PSIA)	PTPLO2	=	14.07 (2,040.00)
CHAMBER PRESSURE, MPa (PSIA)	PC	=	11.72 (1,700.00)
EXIT PRESSURE, kPa (PSIA)	PE	=	41.44 (6.01000)
AREA RATIO	EPSILN	=	26.1992
THROAT DIAMETER, cm (IN.)	DT	=	45.71 (17.9949
NOZZLE EXIT DIAMETER, cm (IN.)	DE	=	23.95 (92.1075
COMBUSTION MIXTURE RATIO	CMR	=	2.60000
LIQUID/SOLID MIXTURE RATIO	XLSMR	=	1.90000
COMBUSTION TEMPERATURE, °K (°R)	TC	=	3687 (6637.00)
RATIO OF SPECIFIC HEAT	G	=	1.20370
GAS MOLECULAR WEIGHT, kg/MOL (LBM/MOL)	XMW	=	11.495 (25.3420)
TOTAL GAS FLOW, kg/SEC (LBM/SEC)	FLOWT	=	4714.64 (10,394.0)
THEORETICAL VAC ISP., SEC	THVISP	=	318.282
THEORETICAL CSTAR, m/SEC (FPS)	CSTAR	=	1694.6 (5,559.64)
CSTAR EFFICIENCY	ETACST	=	0.985000
CF EFFICIENCY	ETACF	=	0.987000
CYCLE EFFICIENCY	ETACYC	=	0.978000
DELIVERED ISPSL, SEC	DISPSL	=	265.154
DELIVERED ISPVAC, SEC	DISPVC	=	302.931
LO2 DENSITY, kg/m^3 (LBM/CF)	DENLO2	=	1140.5 (71.2000)
FUEL DENSITY, kg/m^3 (LBM/CF)	DENF	=	1097.1 (68.4900)
BULK DENSITY, kg/m^3 (LBM/CF)	BULKDY	=	1125.1 (70.2416)
BULK SPECIFIC GRAVITY	BULKSG	=	1.12567
BSG.ISPVAC, SEC	BSISPV	=	340.999

The Second Page Shows HRB Weights, Some Inputs and Some Outputs, Thrusts, and Dimensions for the HRB. Note That the Code Predicts Shorter HRBs Than We Design, Because It Assumes a Cylindrical LO_2 Tank Over Its Entire Length

TURBOPUMP FED (TF) HRB FOR STS PRELIMINARY DESIGN (Cont.)

OXIDIZER WEIGHT, kg (LBM)	WO	=	313926.29 (692,089.)
OXIDIZER VOLUME, m^3 (CF)	VO	=	275.25 (9,720.35)
OXIDIZER ULLAGE FRACTION	OUF	=	0.960000
OXIDIZER TANK VOLUME, m^3 (CF)	VOT	=	286.72 (10,125.4)
OXIDIZER TANK WEIGHT, kg (LBM)	WOT	=	12345.42 (27,217.0)
FUEL WEIGHT, kg (LBM)	WF	=	165224.20 (364,257.)
FUEL VOLUME, m^3 (CF)	VF	=	150.6 (5,318.40)
SOLID FRACTION	FUF	=	0.890000
SOLID CASE VOLUME, m^3 (CF)	VFT	=	169.2 (5,975.73)
SOLID CASE WEIGHT, kg (LBM)	WFT	=	24856.68 (54,799.6)
STRUCTURAL WGT., kg (LBM)	WSTRCT	=	4208.43 (9,278.00)
TCA/TPA WEIGHT, kg (LBM)	WENG	=	8346.74 (18,401.4)
PRESSURIZATION SYSTEM WGT, kg (LBM)	WPRESS	=	1182.52 (2,607.00)
LO2 FEED SYSTEM WGT, kg (LBM)	WOFEED	=	813.75 (1,794.00)
TVC SYSTEM WGT., kg (LBM)	WTVCS	=	1360.78 (3,000.00)
THERMAL PROTECT. SYS. WGT., kg (LBM)	WTPS	=	426.38 (940.000)
SEPARATION SYSTEM WGT., kg (LBM)	WSEPS	=	616.89 (1,360.00)
AVIONICS & ELECTRICAL WGT., kg (LBM)	WAVE	=	1437.89 (3,170.00)
RANGE SAFETY SYS. WGT., kg (LBM)	WRSS	=	68.04 (150.000)
I/F ATTACH HDW. WGT., kg (LBM)	WIFHDW	=	598.74 (1,320.00)
CONTINGENCY (10%) WGT., kg (LBM)	WCONTY	=	5626.22 (12,403.7)
HRB TOTAL DRY WGT., kg (LBM)	WDRY	=	61888.60 (136,441.)
EXPENDED NPCs WGT., kg (LBM)	WXNPCS	=	226.80 (500.000)
RESIDUALS & UNUSEABLES WGT., kg (LBM)	WRESUN	=	6612.29 (14,577.6)
IMPULSE PROPELLANTS WGT., kg (LBM)	WP	=	479152.30 (0.105635E+07)
BOOSTER LIFT-OFF WGT., kg (LBM)	BLOW	=	548973.77 (0.121028E+07)
SEPARATION (BURNOUT) WGT., kg (LBM)	WBOUT	=	69596.49 (153,434.)
SL THRUST, MN (LBF)	FSL	=	11.46 (0.275697E+07)
VAC THRUST, MN (LBF)	FVAC	=	14.01 (0.314865E+07)
HRB PROPELLANT MASS FRACTION [HRB PROPELLANT/BLOW]	FRACT	=	0.872811
BOOSTER DIAMETER, m (FT)	D	=	3.81 (12.5000)
TCA LENGTH, m (FT)	XTCA	=	3.97 (13.0134)
LO2 TANK LENGTH, m (FT)	XOT	=	26.42 (86.6754)
SOLID GRAIN CASE LENGTH, m (FT)	XGR	=	16.11 (52.8612)
TOTAL BOOSTER LENGTH, m (FT) [LENGTH FROM UPPER DOME LO2 TANK TO NOZZLE EXIT PLANE]	XHRB	=	46.50 (152.550)

The Third Page Shows STS Data Including GLOW, Liftoff Thrust/Weight, and Payload Delivered to LEO. Ideal Velocities Are Shown and Accelerations Without Throttling Along With Other STS Weights. The Last Entry (Payload FOM) Is the Payload/GLOW Ratio for the STS

VEHICLE PERFORMANCE FOR TF HRB FOR STS

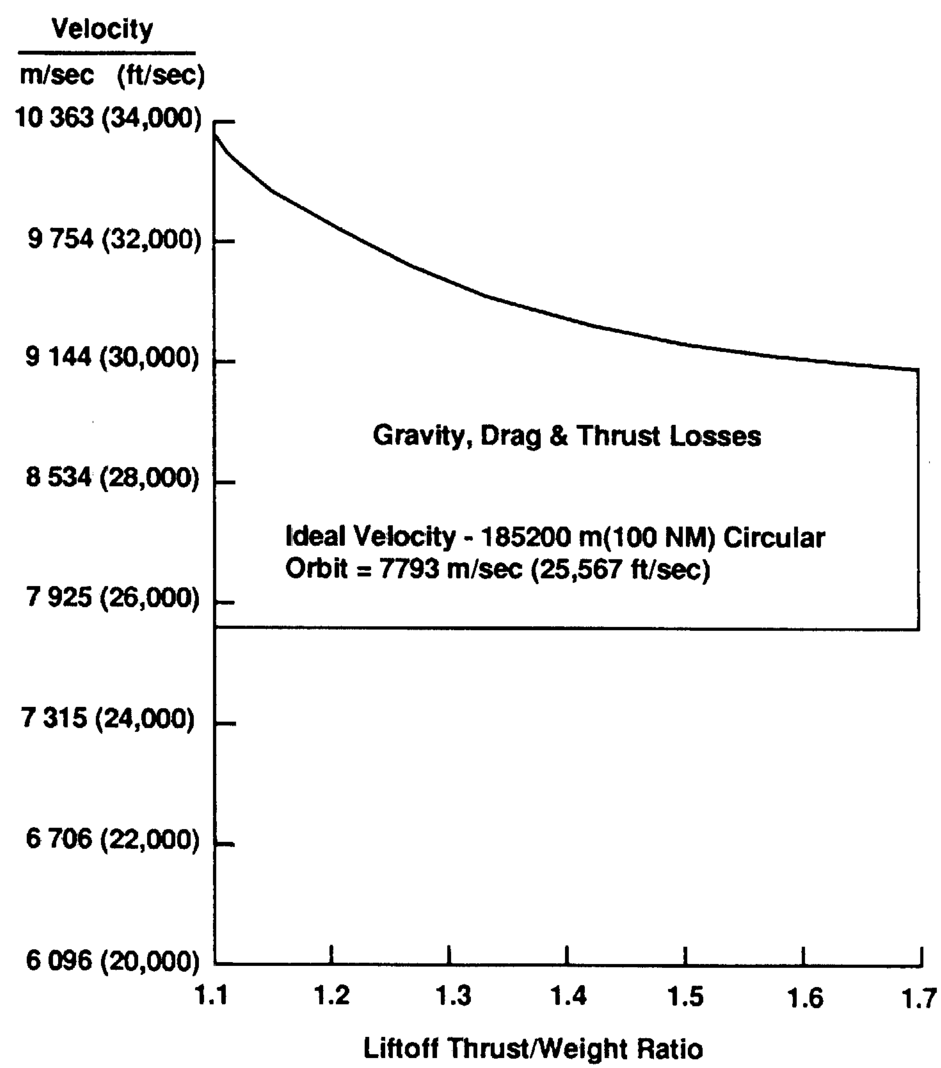
TURBOPUMP FED [TF] HRB, 8-9-89,
CASE: 8 D EBBC - 1,4 TCAs, Pe = 41.37 kPa (6.0 psia),
PROPELLANT: GRAIN #8 SAT. HC [PEBC] + LO2

ORBITER EMPTY WEIGHT, kg (LBM)	WORBTR	=	68439.83 (150,884.)
EXTERNAL TANK BO WGT, kg (LBM)	WETBO	=	30514.07 (67,272.0)
ET PROPELLANT WGT., kg (LBM)	WETPPT	=	720286.54 (0.158796E+07)
ET LIFT OFF WEIGHT, kg (LBM)	WETTLO	=	751031.03 (0.165574E+07)
GROSS LIFT OFF WEIGHT, kg (LBM)	GLOW	=	1950107.00 (0.429925E+07)
THRUST-TO-WEIGHT @ LIFT-OFF	TWLO	=	1.54421
TOTAL VACUUM THRUST, MN (LBF)	FV	=	28.51 (0.641000E+07)
BOOST VELOCITY INCREMENT, m/SEC (FPS)	DELTAV	=	2813.58 (9,230.89)
TOTAL EFFECTIVE ISP, SEC	TISPVC	=	326.672
BOOST ET PROP DEPLETION, kg (LBM)	WETDPL	=	180955.69 (398,939)
BOOST PROPS DEPLETION, kg (LBM)	WPROPS	=	1139256.20 (0.251163E+07)
WEIGHT BEFORE SEPARATION, kg (LBM)	W2	=	810846.26 (0.178761E+07)
POST SEPARATION WGT., kg (LBM)	W3	=	671656.90 (0.148075E+07)
MECO WEIGHT, kg (LBM)	W4	=	131641.12 (290219.)
SSMES VELOCITY INCREMENT, m/SEC (FPS)	DELT34	=	7235.9 (23,739.8)
TOTAL VELOCITY INCREMENT, m/SEC (FPS)	TELTAV	=	9362.4 (30,716.7)
CHECK VELOCITY INCREMENT, m/SEC (FPS)	ZELTAV	=	9362.4 (30,716.7)
CHECK VALUE FOR RATIO	CRATIO	=	0.167617E-01
STS PROPELLANT MASS FRACTION	XMU	=	0.594162
TIME TO SEPARATION, SEC	T	=	128.000
DISTANCE TO SEPARATION, m (FT)	PATH	=	154070.3 (505,460.)
ACCELERATION AT SEPARATION, m/SEC^2 (G)	ACC	=	35.16 (3.58578)
STS PAYLOAD, kg (LBM)	PAYLD	=	32687 (72,062.5)
STS PAYLOAD RELATED FOM	RATIO	=	0.167617E-01

Our Weight Code Iterates to Use the Correct Ideal Velocity Ratio, Using This Relationship for Ideal Velocity Increment vs Thrust/Weight at Liftoff. This Same Algorithm Was Used for NASA, MSFC by Aerojet During Early ALS Engine Studies and Is Included in the AIAA Paper We Prepared About That Work, Indicated on Page 151

GenCorp Aerojet

ΔV MODIFICATION FOR SIMPLIFIED LEO PROPULSION REQUIREMENTS



Velocity Increment, ft/sec ≡ ZELTAV
ZELTAV = 1000* (28.69258 - 2.9834398/TWLO + 9.4337118/(TWLO**2))

3.21.0.8

Martin Marietta Flew Our Design Point HRB to LEO Using Their Trajectory Code With Varying Weight STS Orbiters. The Resulting Figure Shows Payload Results of Approximately 33020 kg (72,800 lbm) to LEO and Corresponds to an Orbiter Weight of About 96165 kg (212,000 lbm), or Only 2268 kg (5,000 lbm) Less Than the Magellan Orbiter Weight

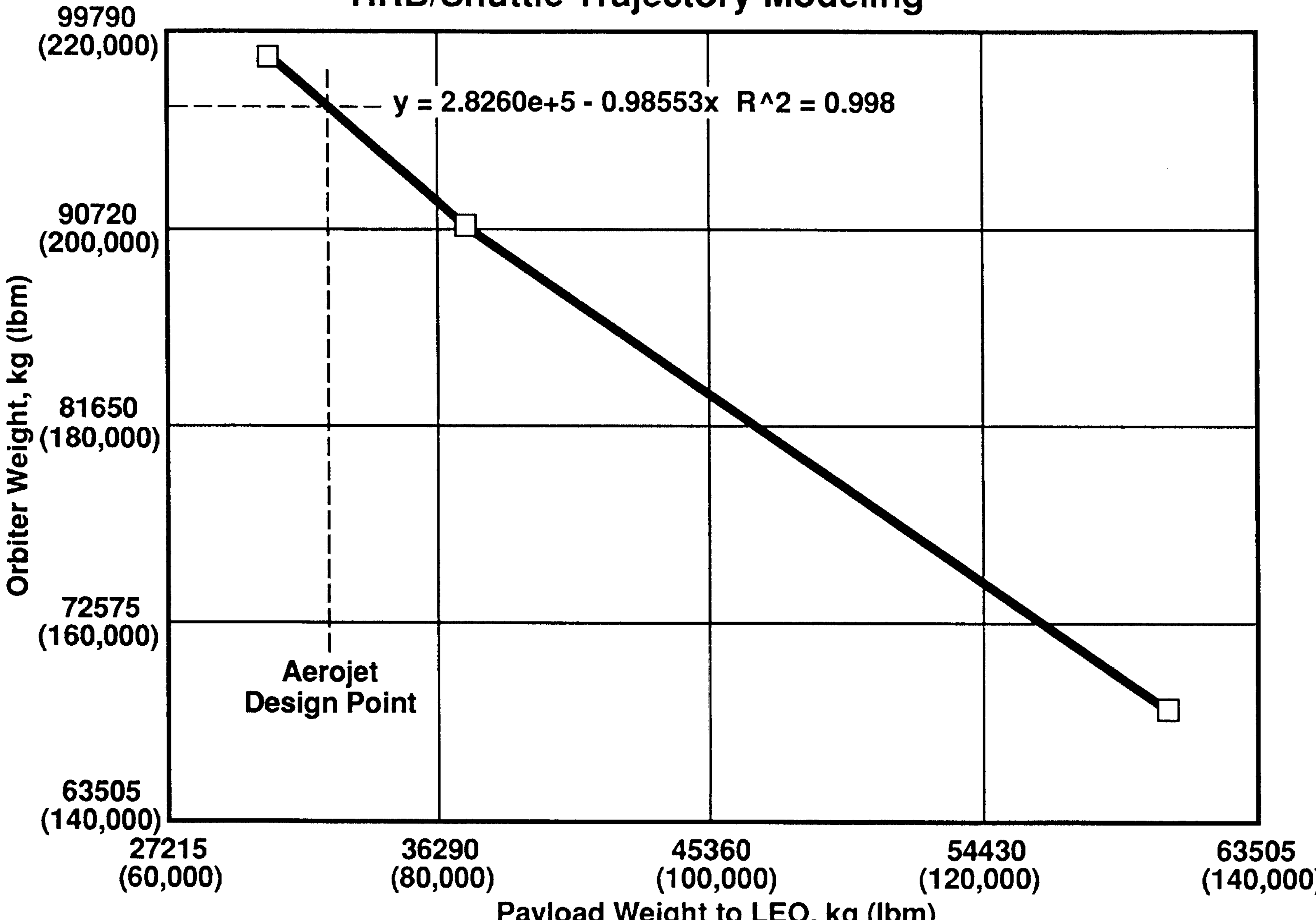
C-3

Our Weight Results Depended, In Part, on Having a Baseline Propulsion System to Provide Fixed Inputs to Our Weight Model Code. Aerojet and Martin Marietta Collaborated to Collect a Coherent Data Set

GenCorp Aerojet

OUR WEIGHT STUDY RESULTS ARE REASONABLE

HRB/Shuttle Trajectory Modeling



M13 HD-067

DEFINE BASELINE HYBRID PROPULSION DESIGN AND OPERATING POINT

We Selected Our Baseline Operating Point From SRB and LRB for STS Data. SRBs and Pressure Fed HRBs Should Optimize at About the Same Pressure, or at About 6.9 MPa (1,000 psia) Combustor Pressure. Pump Fed Bipropellant Boosters Optimize at About 17.2 MPa (2,500 psia.) Therefore It Appeared That a Pumped HRB Would Have Maximum Performance at About 12.4 MPa (1,800 psi) Combustor Pressure. Extra Solid Case Hoop Stress Can Be Taken by Hoop Wrapped Graphite Fibers, So the Same Basic Steel Case Can Be Used to Provide Stiffness for Launch Loads.

We Selected the ASRM Diameter of 3.81 m (150 in.), and Evaluated Two Nozzle Diameters With One Nozzle Per Large HRB, as the SRBs Have. One Nozzle Fits the Current MLP, 3.80 m (149.5 in.) dia, and the Larger Diameter Nozzle Gives Better Performance With 41.37 kPa (6 psia) Exit Pressure

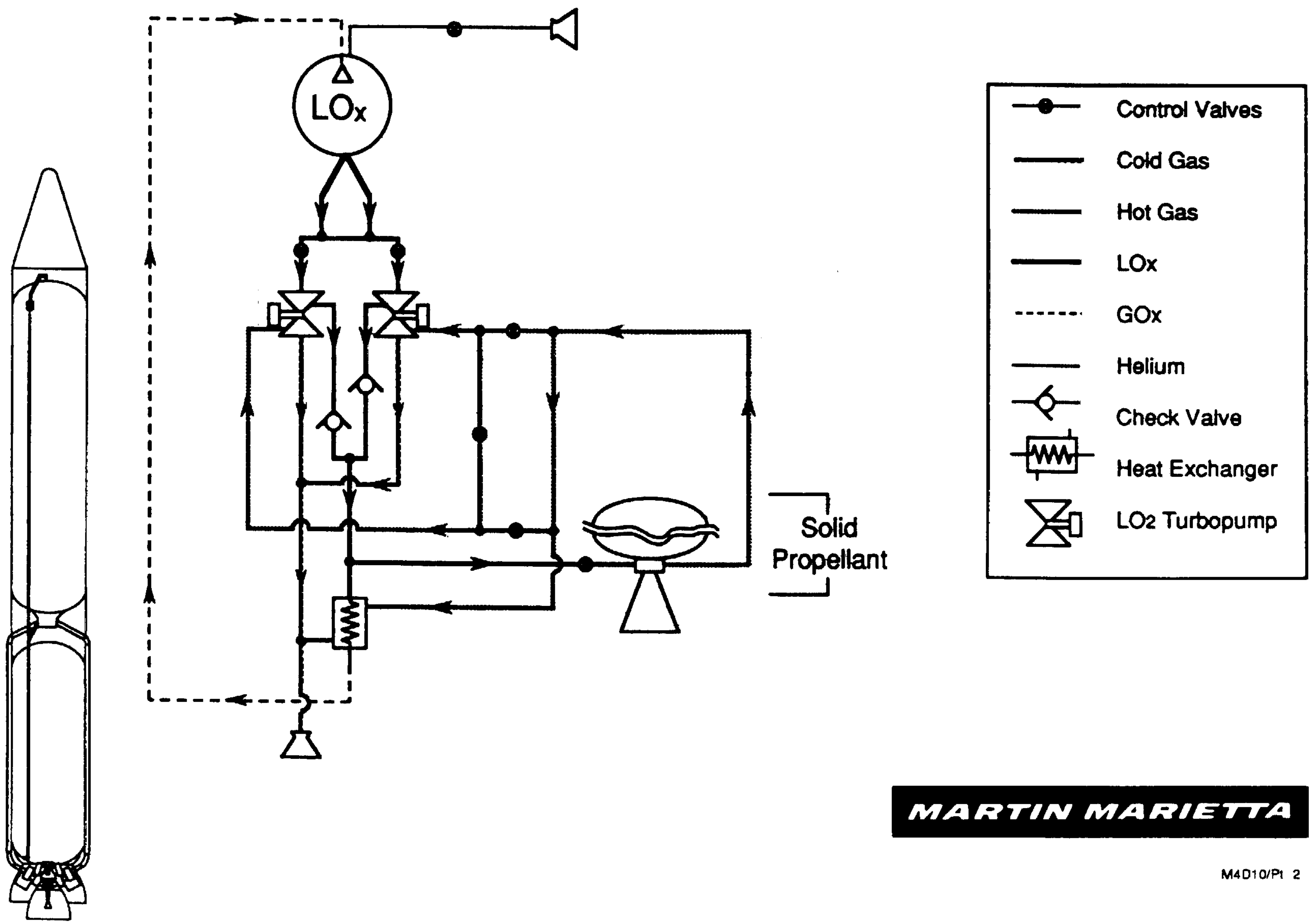
WE DETERMINED LARGE HRB BASELINE OPERATING POINT DESIGN DATA

Parameters	Pump Fed	Pressure Fed
P_c, MPa (psia)	12.40 (1,800)	6.89 (1,000)
P_e, kPa (psia) at ε	41.37 (6.0) at 26.2	41.37 (6.0) at 12.3
D_e, m (in.)	4.70 (185)	5.54 (218)
and		
P_e, kPa (psia) at ε	73.78 (10.7) at 17.4	90.32 (13.1) at 9.5
D_e, m (in.)	3.80 (149.5)	3.80 (149.5)
D_{HRB}, m (in.)	3.81 (150)	3.81 (150)

Martin-Marietta Aerospace Group (Denver) Supplied Us With Baseline Designs for Both Pump and Pressure Fed HRBs. The Pump Fed Version Uses Autogenous Tank Pressurization (GO_2)

STAGED COMBUSTION HYBRID—PUMP FED

Autogenous Pressurization



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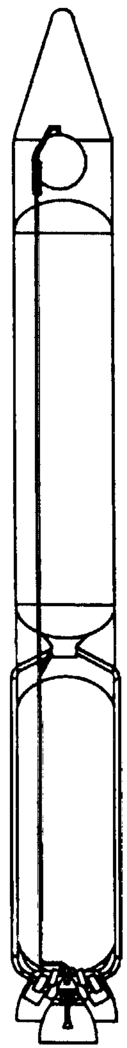
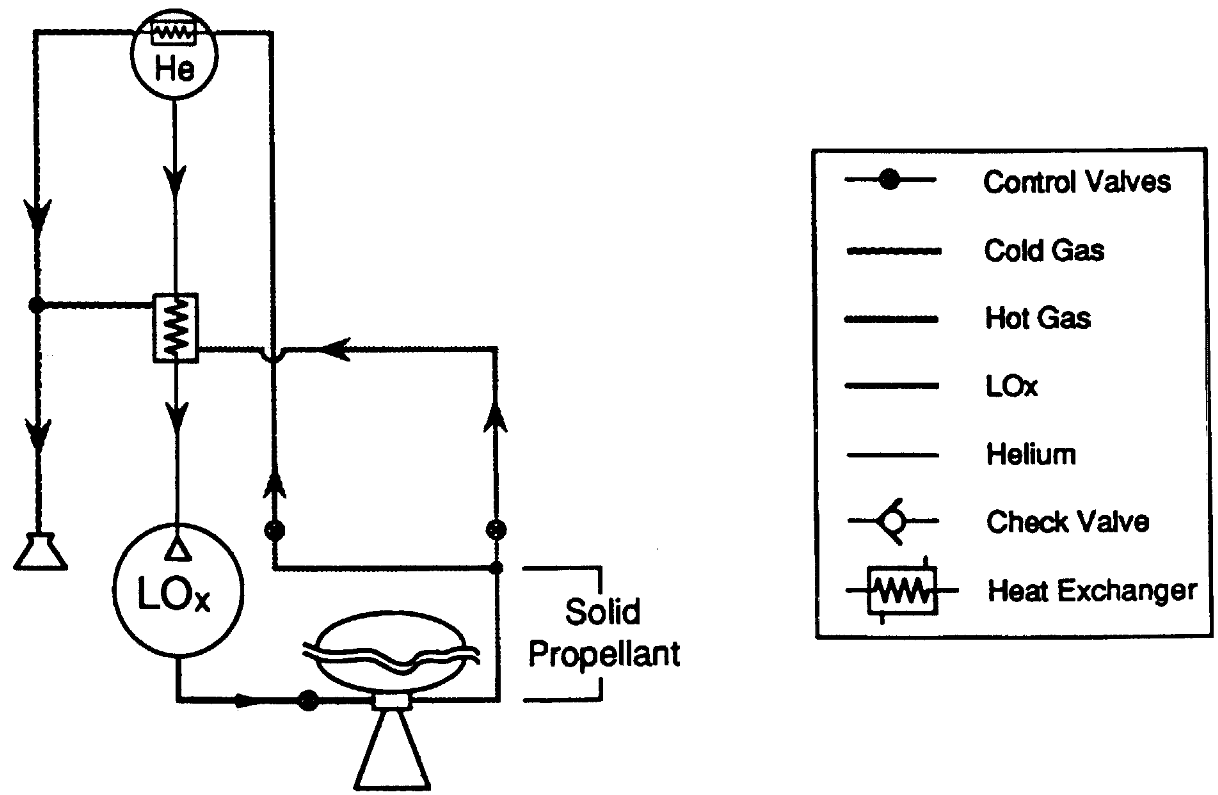
M4D10/Pt 2

The Pressure Fed Baseline HRB Uses Solid Case Heated, Supercritical Helium for LO_2 Tank Pressurization, a Concept Validated by Our Screening Studies. One Heat Exchanger Heats Helium on Its Way to the LO_2 Tank, and the Other Improves Helium Bottle Gas Utilization

STAGED COMBUSTION HYBRID—PRESSURE FED

Supercritical He Pressurization





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M4D10/P1 2

These Are Weights Baselined for the SLSC HRBs

SOLID/LIQUID STAGED COMBUSTION BASELINE CONFIGURATION WEIGHTS*

	Staged Combustion Configuration	
	Pump	Pressure
Structure:		
Nose Cone Weight	706 (1,557)	750 (1,653)
Cross Beam/Forward Skirt Weight	1603 (3,533)	3694 (8,144)
LO_x/HRM Interstage Weight	1900 (4,188)	2530 (5,577)
LO_x Tank Weight		
TVC (Thrust Vector Control)	1057 (2,330)	1057 (2,330)
TPS (Thermal Protection System)	426 (940)	458 (1,010)
Separation System	617 (1,360)	617 (1,360)
Range Safety System	68 (150)	68 (150)
I/F Attach Hardware	599 (1,320)	599 (1,320)
LO_x Feed System	814 (1,794)	1502 (3,311)

	Staged Combustion Configuration	
	Pump	Pressure
Avionics & Electrical System	1438 (3,170)	1429 (3,150)
Pressurization System:		
Pressurant (Helium)	N/A	1512 (3,334)
Tankage & Insulation	N/A	3101 (6,836)
Feedlines & Valves	N/A	309 (681)
Heat Exchangers (two)	N/A	569 (1,255)
LO_x Required	N/A	1542 (3,400
Autogenous Heat Exchanger	236 (520)	N/A
Feedlines	908 (2,002)	N/A
Conduits & Fairings	249 (548)	N/A
Fill & Vent Valves	26 (57)	N/A
LO_x Required	1382 (3,047)	N/A
Contingency (10%)	1794 (3,955)	3831 (8,447)

*Weights in kg (lbm)

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Here Is How They Compared With Corresponding LRB Baselines at the STS Level

HYBRID ROCKET BOOSTER PERFORMANCE FOR THE NSTS

	HRB Pump-Fed S.C.	HRB Press-Fed S.C.	LRB Pump-Fed	LRB Press-Fed
Propellants	LOX/Solid	LOX/Solid	LOX/RP	LOX/RP
Gross Liftoff Weight, Metric Tons (lb)	1798 (3,964,170)	1983 Q(4,372,400)	1894 (4,175,938)	2116 (4,664,931)
Jettison Weight, Metric Tons (lb) 2 Boosters	123 (271,176)	174 (384,104)	123 (271,304)	215 (473,618)
Impulse Propellant Weight, Metric Tons (lb) 2 Boosters	794 (1,751,304)	924 (2,037,350)	889 (1,959,086)	1019 (2,245,410)
Payload, Metric Tons (lb)	32.3 (71,215)	32.2 (70,996)	32.9 (72,499)	33.0 (72,853)
Maximum Vacuum Thrust, MN (lb) Booster	12.87 (2,892,510)	13.93 (3,132,050)	11.09 [2,494 k (NPL)]	11.96 [2,688 k (NPL)]
Liftoff Thrust/Weight	1.528	1.442	1.253	1.524
Maximum Percent Throttle	70	69	65	NPL
Total Impulse Nm-sec, (lb-sec) x 10^{-6}	12.54 (2.82)	14.23 (3.2)	————	————
Qmax, kg/m^2 (psf)	3466 (710)	3486 (714)	3427 (702)	3466 (710)
Burn Time, sec	133	135	131.8	123.7
Booster Length, m (ft)	38.1 (125)	38.1 (125)	45.5 (149.4)	49.6 (162.7)
Booster Diameter, m (ft)	3.81 (12.5)	4.05 (13.3)	4.65 (15.25)	4.94 (16.2)
Engine Isp (vac/s) sec	322/287	318/285	322/265 (NPL)	317/253 (NPL)
Engine O/F	1.0	1.0	2.6 (NPL)	2.67
Engine Area Ratio	21.2	11.5	21.2	11.7
Engine Chamber Pressure, MPa (psia)	12.41 (1800)	6.89 (1000)	7.12 [1033 (NPL)]	3.44 [499 (NPL)]

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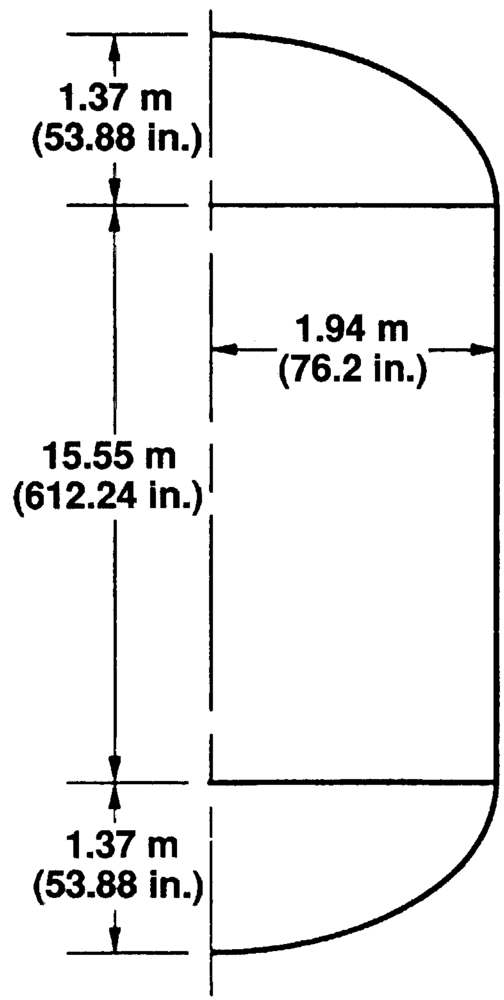
Baselined Pressure and Pump Fed HRB LO_2 Tanks Were Studied by Martin Marietta

HRB LOX TANK CONFIGURATION



Tank Sizing Assumptions:

- Gravitational Acceleration: 13.73 m/sec^2 (1.4g) on Full Tank (Lift Off)
- Gravitational Acceleration: 29.42 m/sec^2 (3.0g) on 1/2 Full Tank (Maximum Q)
- Hydro Test With Water at Limit

Pump Fed:

- Tank Ullage Pressure 345 kPa (50 psi) (Limit)

Pressure Fed:

- Tank Ullage Pressure 8963 kPa (1300 psi) (Limit)

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They Evaluated Four Concepts, AlLi and Stainless Steel With and Without Glass/Epoxy Overwrap. For the Pump Fed System AlLi Was the Best (With or Without Overwrap) and for the Pressure Fed System, Overwrapping Was Best (With AlLi or Stainless Steel).

This Reinforced the Requirement for Overwrapped Steel for the Solid Case

HRB LOX TANK

Trade Study Summary

1. Four Configurations

   - Weldalite™ 049 Shell with S-Glass/ Epoxy Overwrap
   - Weldalite™ 049 Shell
   - Stainless Steel 304L with S-Glass/Epoxy Overwrap
   - Stainless Steel 304L Shell

2. Results (Weight Criterion)

   - Pump Fed System - Lowest Weight: Overwraped Wedalite™
     - Close Second: Weldalite Shell

   - Pressure Fed System - Lowest Weight: Overwraped Stainless Steel
     - Close Second: Overwraped Weldalite™

3. Manufacturing Cost Should be Included in Separate Trade Study with Consideration of Attachment Loads.

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This Chart Provides Detailed Weight Results for the Pump Fed System. Note That Stainless Steel by Itself Is Heaviest

HRB LOX TANK

Configuration Trade Study Result - Pump Fed System				
	Wall Thickness, cm (In.)			
Configuration	Upper Dome	Lower Dome	Barrel	Weight, kg (lb)
Weldalite™ 049	0.051 (0.020)	0.051 (0.020)	0.051 (0.020)	1426 (3,143)
With S-Glass/Epoxy Overwrap	0.089 (0.035)	0.140 (0.055)	0.213 (0.084)	
Weldalite™ 049	0.148 (0.057)	0.201 (0.079)	0.279 (0.110)	1612 (3,554)
Stainless Steel 304L	0.051 (0.020)	0.051 (0.020)	0.051 (0.020)	1885 (4,156)
With S-Glass/Epoxy Overwrap	0.076 (0.030)	0.122 (0.048)	0.183 (0.072)	
Stainless Steel 304L	0.140 (0.055)	0.193 (0.076)	0.272 (0.107)	4636 (10,220)
Common Features:	Elliptical Domes; External Thermal Protection Material, SOFI (Not Included in Weight), Tank Pressure 345 kPa (50 psi)			

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This Chart Provides Pressure Fed System Tank Details. Stainless Steel by Itself Again Is Heaviest , Only More So Than With Pump Fed HRBs

HRB LOX TANK

Configuration Trade Study Result - Pressure Fed System				
	Wall Thickness, cm (In.)			
Configuration	Upper Dome	Lower Dome	Barrel	Weight, kg (lb)
Weldalite™ 049	0.051 (0.020)	0.051 (0.020)	0.051 (0.020)	20433 (45,048)
With S-Glass/Epoxy Overwrap	2.59 (1.02)	2.64 (1.04)	3.73 (1.47)	
Weldalite™ 049	2.8 (1.105)	2.86 (1.127)	4.04 (1.591)	23668 (52,180)
Stainless Steel 304L	0.051 (0.020)	0.051 (0.020)	0.051 (0.020)	18417 (40,602)
With S-Glass/Epoxy Overwrap	2.26 (0.89)	2.29 (0.90)	3.25 (1.28)	
Stainless Steel 304L	2.69 (1.060)	2.75 (1.081)	3.88 (1.527)	67217 (148,190)
Common Features: Elliptical Domes; External Thermal Protection Material, SOFI (Not Included in Weight), Tank Pressure 8963 kPa (1,300 psi)				

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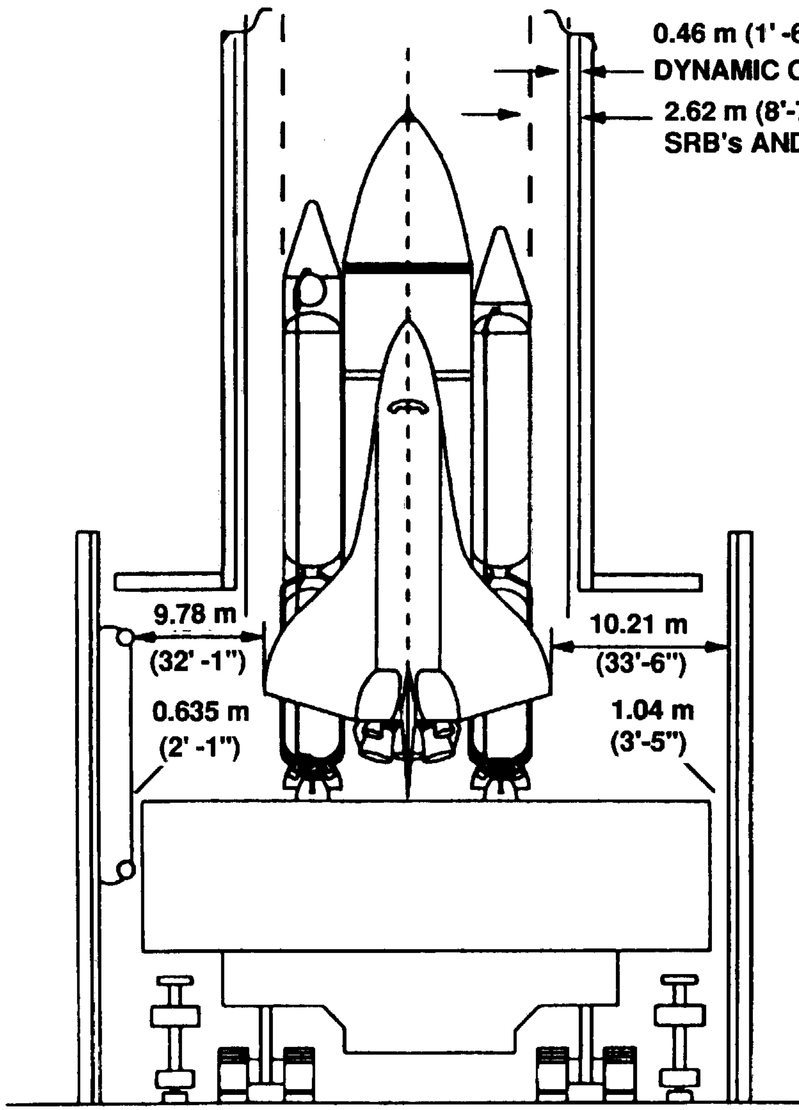
FACILITIES

Martin Marietta Investigated Our Baselined Facilities at the JFK Spaceflight Center

There Is Plenty of Clearance for Any HRB Concept in the VAB (Next Two Pages)

VAB INTEGRATION CELL DOOR CLEARANCES



DIMENSIONS	M	[In. (FT)]
ET	8.41	[331" (27'7')]
ET/HRB SEP	0.3	[12" (1'-0")]
TOTAL DOOR WIDTH	22.1	[871.5" (72'-7.5")]
SRB/DOOR SEP	2.62	[103" (8'-7")]
SRB DIAMETER	3.71	[146" (12'-2")]
STANDARD MIN. CLEARANCE (DYNAMIC)	0.46	[18" (1'-6")]
MAX. HRB DIA. DYNAMIC CLEARANCE	6.32	[249" (20'-9")]
MAX. HRB DIA. W/DYNAMIC CLEARANCE	5.87	[231" (19'-3")]

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M10/D5/Hybrid-3

VAB INTEGRATION CELL DOOR CLEARANCE

- VAB High Bay 1/3 Exit Door Opening Is Approximately 22.1 m [870 in. (73 ft)]
- Current STS Clearance Between Edge of Door and SRB Is 2.62 m (103 in.)
- Current STS Dynamic Minimum Clearance Requirements (Shuttle on MLP and Under Motion) and Any Portion of Facility Is 0.46 m [18 in. (1 ft 6 in.)]
- Hybrid Rocket Booster Could Grow to 5.87 m (231 in.) Diameter and Still Meet Dynamic Clearance Requirements

MARTIN MARIETTA

Crane Lift Capacity Is More Than Adequate for the Single Segment Solid Case Loaded With Propellant. The Crane Modified for ASRM Will Lift the Entire Large HRB Without LO_2

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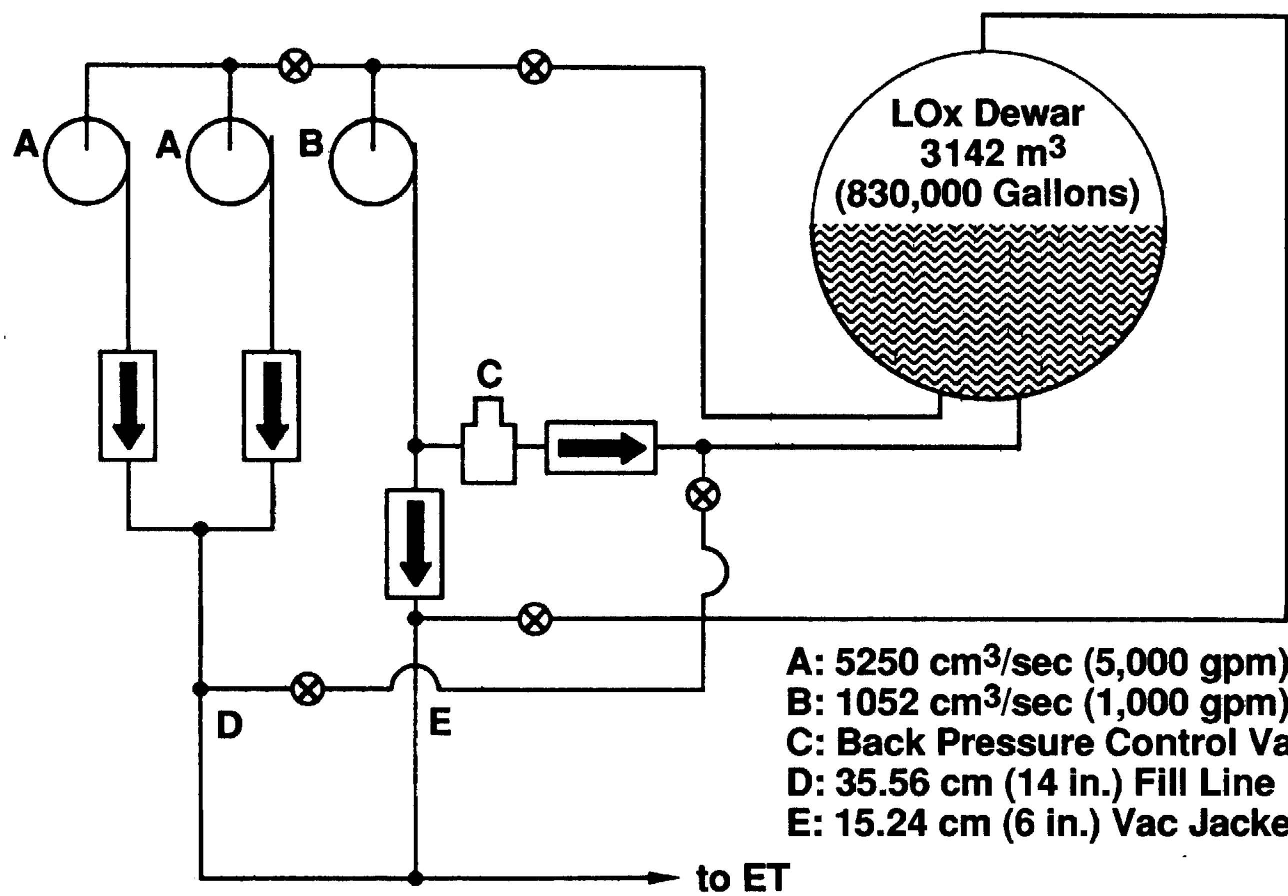
VAB LIFT CAPABILITY

	Metric Tons (U.S. Tons)
• Current	227 (250)
	5.6 to 1 Safety Factor
• Modified for ASRM	272 (300)
• Required for HRB	
(Solid Case With Grain)	<227, ~209 (<250, ~230)
(HRB Without LO_2)	<272, ~236 (<300, ~260)

The Existing LO_2 System Is Adequate for HRB Needs (Next Two Pages)

EXISTING STS LOX SERVICING SYSTEM



A: 5250 cm^3/sec (5,000 gpm) Pump
B: 1052 cm^3/sec (1,000 gpm) Pump
C: Back Pressure Control Valve
D: 35.56 cm (14 in.) Fill Line
E: 15.24 cm (6 in.) Vac Jacket Replenish Line

MARTIN MARIETTA

LOX SERVICING

- LOX Loading System Will Be Similar to That Used for STS
- ET and HRB's Can Be Loaded Concurrently From a Common LOX Storage Dewar
- ET Storage Dewar Is 3142 m^3 (830,000 Gallons) Capacity
- Main Fill Rate Approximately 10500 cm^3/sec (10,000 Gallons/minute) With Two Pumps, ie, 2 pumps x 5250 cm^3/sec (5,000 gallons/minute)
- Replenish Rate Is 1052 cm^3/sec (1,000 Gallons/minute)

MARTIN MARIETTA

OPERATIONS

Martin Marietta Investigated Our HRB Operations at JFK SFC vs SRBs

Qualitatively, HRBs Offer Several Operational Benefits vs SRBs

OPERATIONAL BENEFITS FOR HYBRIDS

- IMPROVED SAFETY
  - Eliminate Catastrophic Scenario for Shuttle (Inert Grain)
  - Reduce Number of Lift Operations
  - No Propellants Involved During Lifting Operations
  - Propellant Loading at Launch Pad

- SIGNIFICANTLY IMPROVED ENVIRONMENTAL EFFECTS
  - No HCL in Exhaust Products

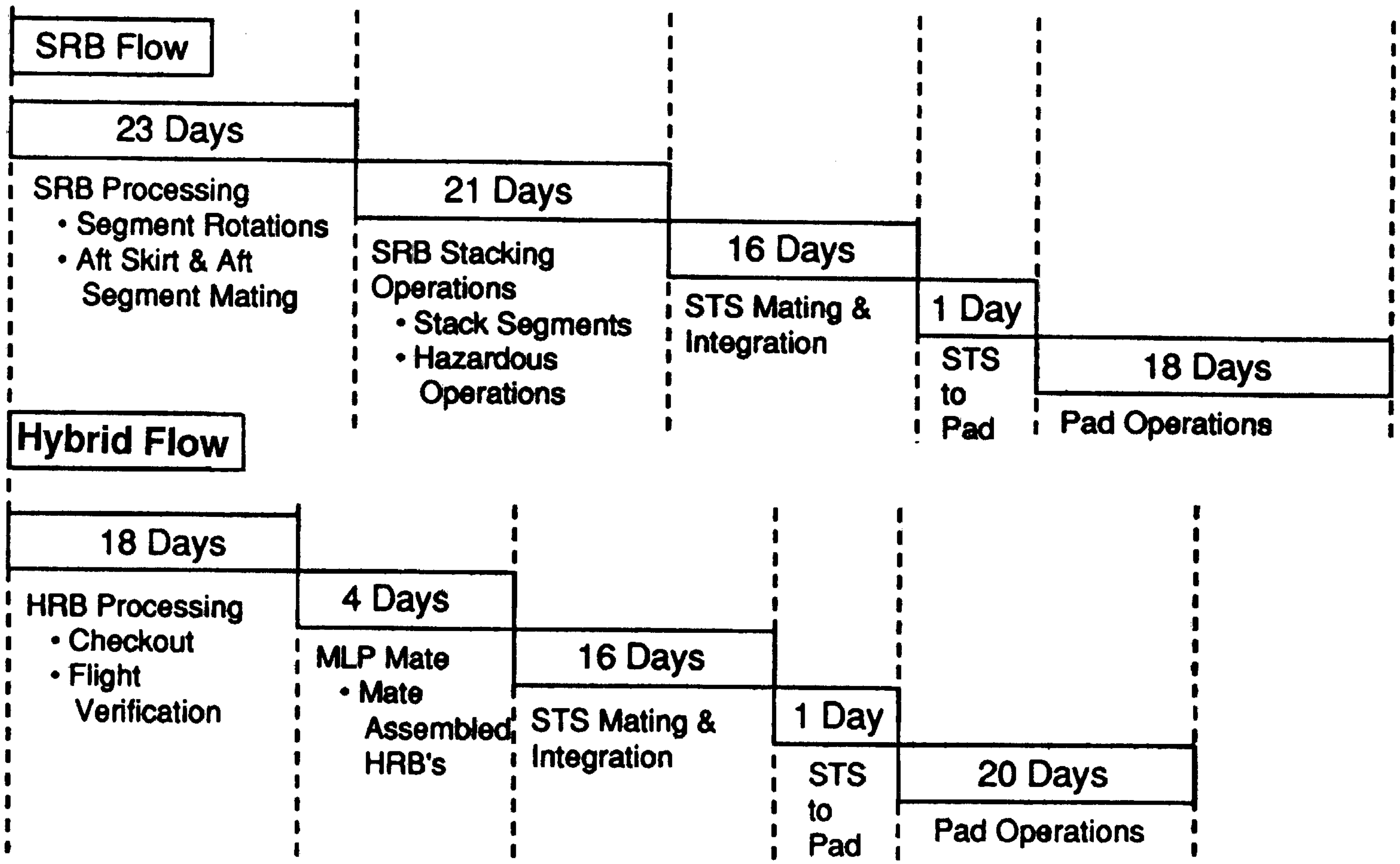
- KSC PREFERRED PROPELLANT (LOx)
  - Commonality with Existing Space Transportation System
  - Existing Support Infrastructure

MARTIN MARIETTA

Quantitatively, 20 Days of Launch Operations Per Launch Could Be Saved With HRBs. Most Savings Accrue From the Lack of SRM Segments in Our HRB

HYBRID LAUNCH OPERATIONS

Based on 1994 NSTS Timeline Estimates



MARTIN MARIETTA

M4D10/PI 2

DEFINE LCC SCENARIOS

The Contract SOW Requires a 2 by 2 by 2 Matrix Study of HRB Use Scenarios, i.e.:

- High vs Low Flight Rate
- Two vs Eight HRBs/Flight
- Expendable vs Reusable HRBs

These Scenarios Resulted in Varying Numbers of HRBs Produced and Flown; and Provides Information Needed for LCC and FOM Score Calculations

EIGHT MISSION MODELS FOR HRB STUDY

	Number of HRB				
	Expendable		Reusable*		
HRB No. / Flt. Rate	2 Per Vehicle	8 Per Vehicle	2 Per Vehicle	8 Per Vehicle	Number of Flights
1/Month	288	1,152	42	168	144
1/Week	1,248	4,992	182	728	624

*Based Upon 10 Uses and 1 Year Refurbishment

DEFINE LCC SCOPE AND DATA BASE

We Considered Five Elements in Our Life Cycle Cost Model. Production Cost Was the Largest, With Payload Second and Development Third

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HRB STUDY
LIFE CYCLE COST ELEMENTS

1. Payload Performance Cost Impact

2. Development Cost

3. Production Cost

4. Operations Cost Impact

5. Facility Cost Impact

Production Costing Began With Estimating First Unit Production Costs. It Is Based on a Cost/per unit weight x Weight Summing Method

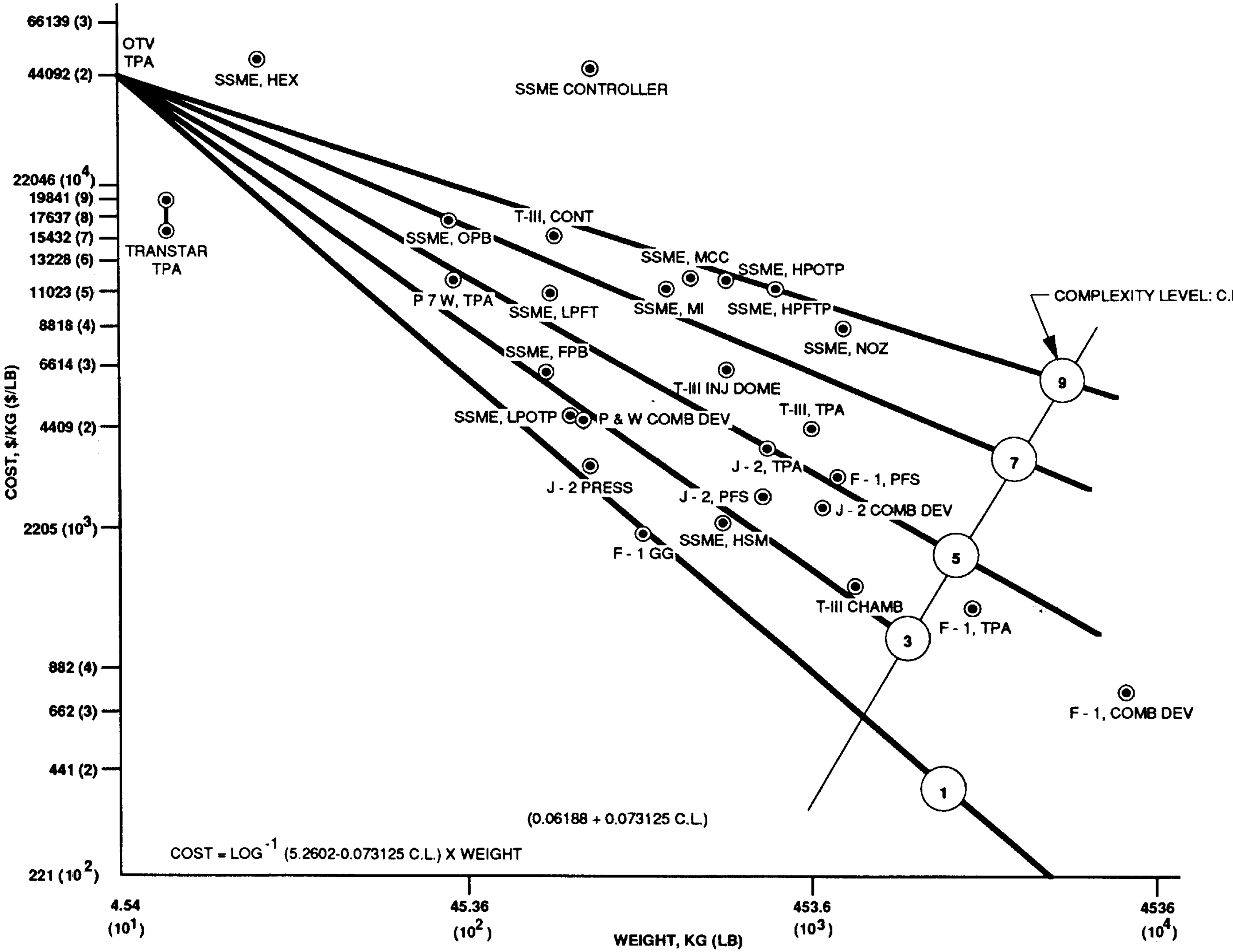
HRB FUPC METHODOLOGY

First Unit Production Cost Prediction

- Major Subsystem Weights Obtained From HRB Performance Trajectory Code
- Subassembly Components Defined by Design Concept Options
- Weight Allocations Used From Solid and Liquid Databases Scale A/R With ELES Code
- Cost vs Weight "Cost Estimating Relationships" (CERs) Derived From Historical Cost Database
- Subassembly Costs Calculated From CERs
- Costs Summed for HRB Stage Total Cost

This Chart Validated the Correlation of Component Cost/Unit Weight With Weight and Complexity by Illustrating How Actual Rocket Engine Components Correlate

COMPONENT COST RELATIONSHIPS



M13 HD-041

For Large Boosters the Cost Per Pound of Components Does Vary Considerably With Weight and Complexity. Note How High Weight, Simple Items Such as Propellant, Tanks, Solid Cases, Liners, etc., Have Low Cost per Unit Weight, Whereas Relatively Small, Low Weight, and Complex Items Such as Turbopumps, Thrust Chambers, and Gimbal and Feed Systems Are More Costly/ per Unit Weight

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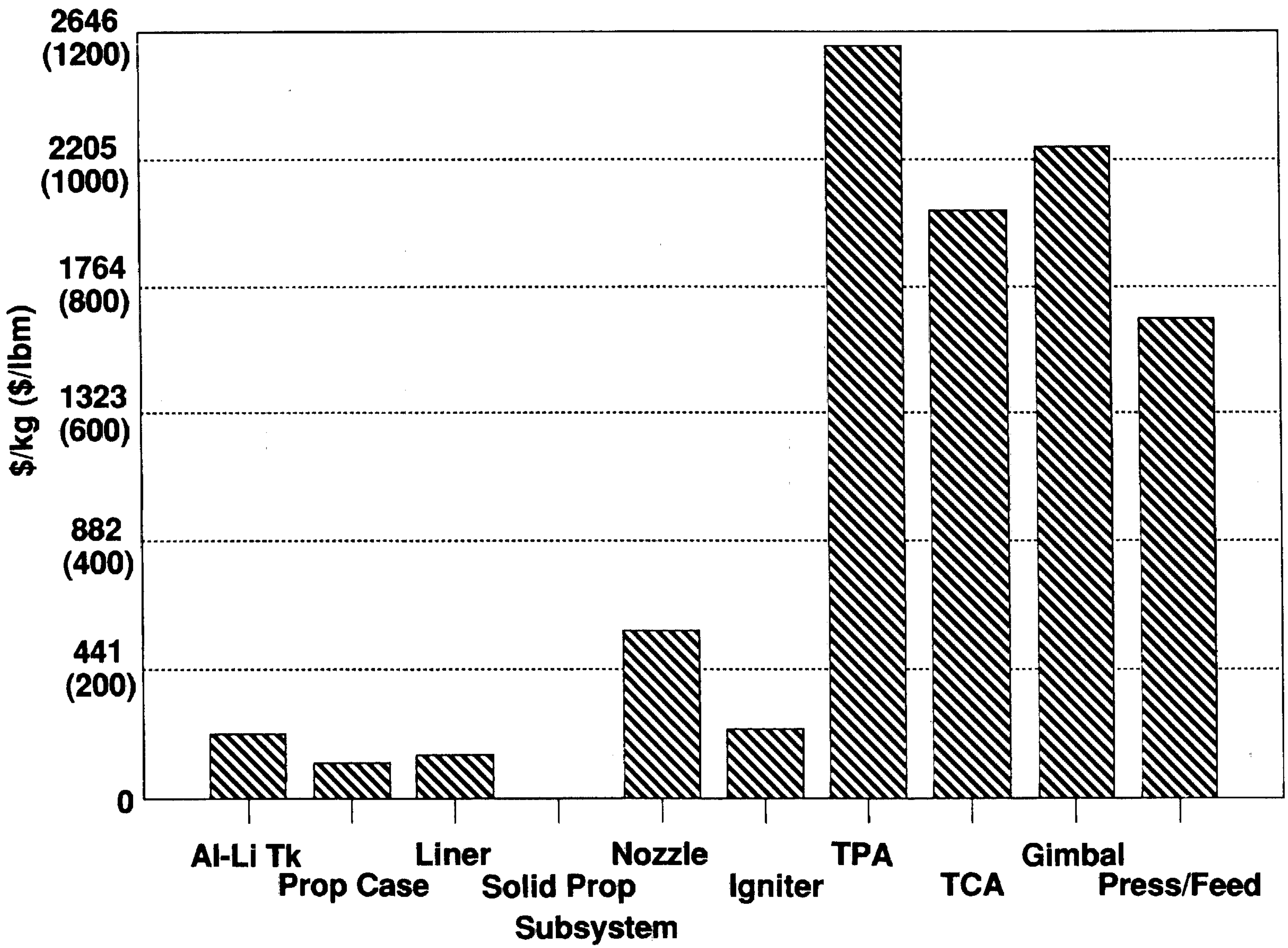
HRB COMPONENT COST FACTORS



The Average Unit Production Cost Determined the Production Cost, and Was Estimated From the First Unit Production Cost With a Learning Curve Slope Analysis Shown. This Graph Assumes the One Flight/Month Scenario

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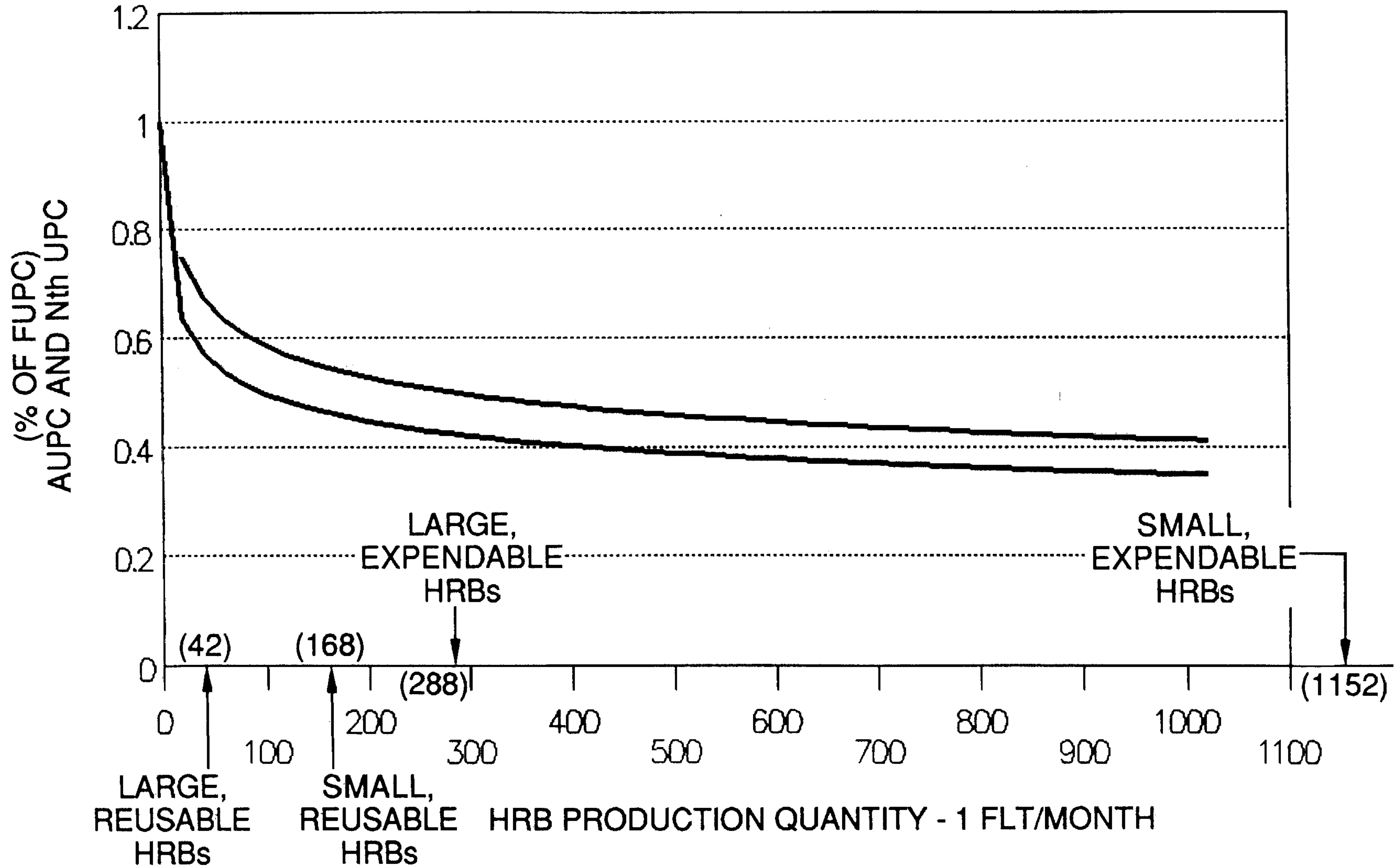
UNIT PRODUCTION COST VS QTY
90% LEARNING CURVE SLOPE



This Graph Was Enlarged to Accommodate the Larger Numbers of HRBs Required for the One Flight/Week Scenario

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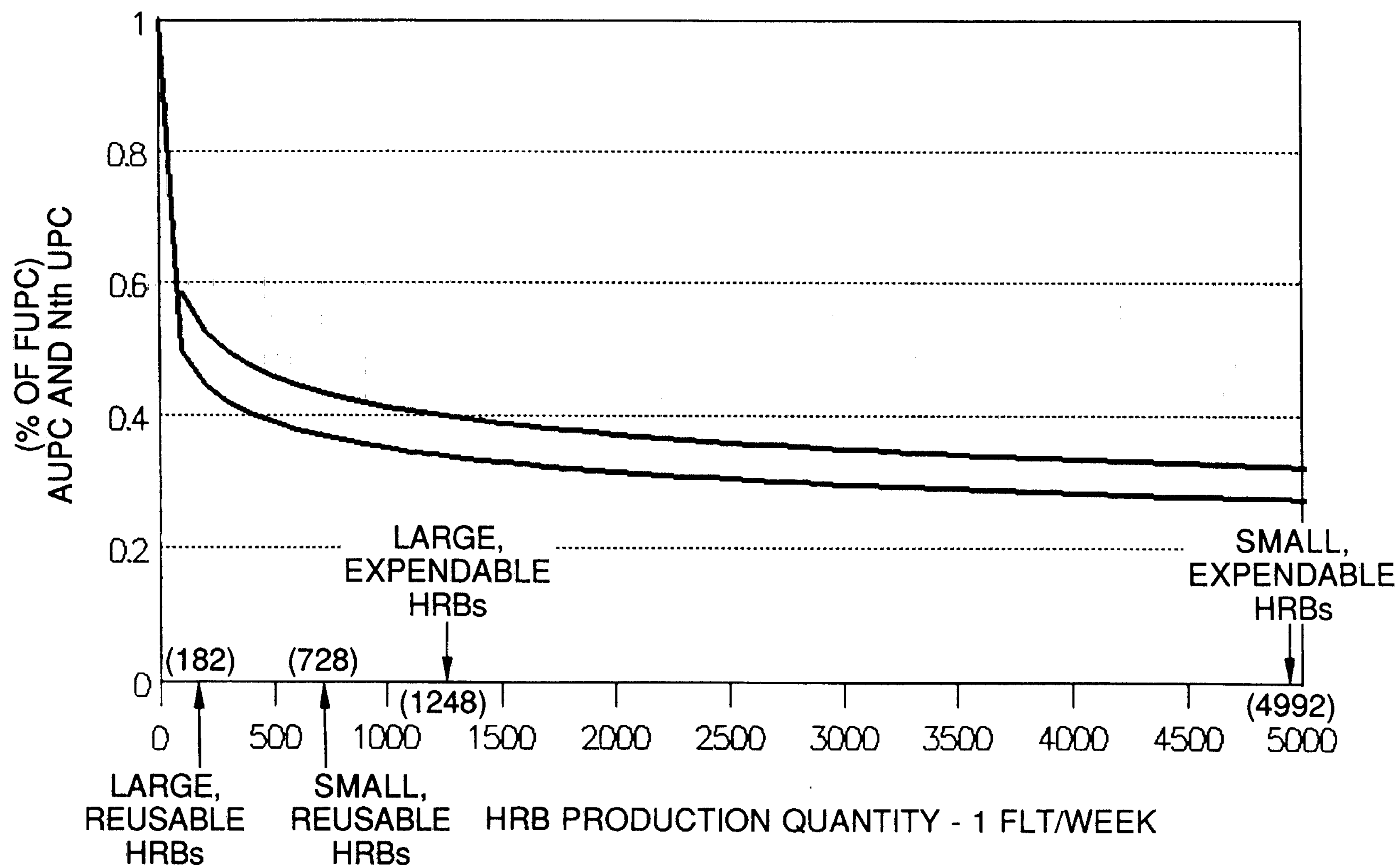
UNIT PRODUCTION COST VS QTY 90% LEARNING CURVE SLOPE



Our Development Cost Methodology Depended Upon Hardware Costs From the Production Cost Analysis, Without Learning Curve Savings Imposed, and With Test Costs Added. To Simplify Our Analyses, We Identified Only the Differences in Test Costs Between Any Two Options, to Obtain the Difference in Development Costs

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HRB DEVELOPMENT COST METHODOLOGY

INPUTS	COMPONENT DEVELOPMENT COST
DESIGN DEFINITION	IDENTIFY COMPONENTS
FUPC	COMPONENT COST
DEVELOPMENT PLAN	COMPONENT DEVEL QUANTITY
ENGR DEVEL DATABASE	TEST COST/ COMPONENT
DEVELOPMENT PLAN	# TESTS/ COMPONENT

HARDWARE COSTS

TESTS COSTS

DEVELOPMENT COST ESTIMATE

In Task 3, Concept Evaluation and Selection, We:

- Developed a Life Cycle Cost and Payload to LEO - Based Scoring Computer Model Referred to as a Figure of Merit (FOM)
- Computed HRB Scores for Each Candidate
- Ranked and Selected Candidates, Selecting Highest Scores Wherever Practical and Possible

We Will Begin by Showing Our Selection Results First, Then Our FOM Model and the Concept Scores It Produced

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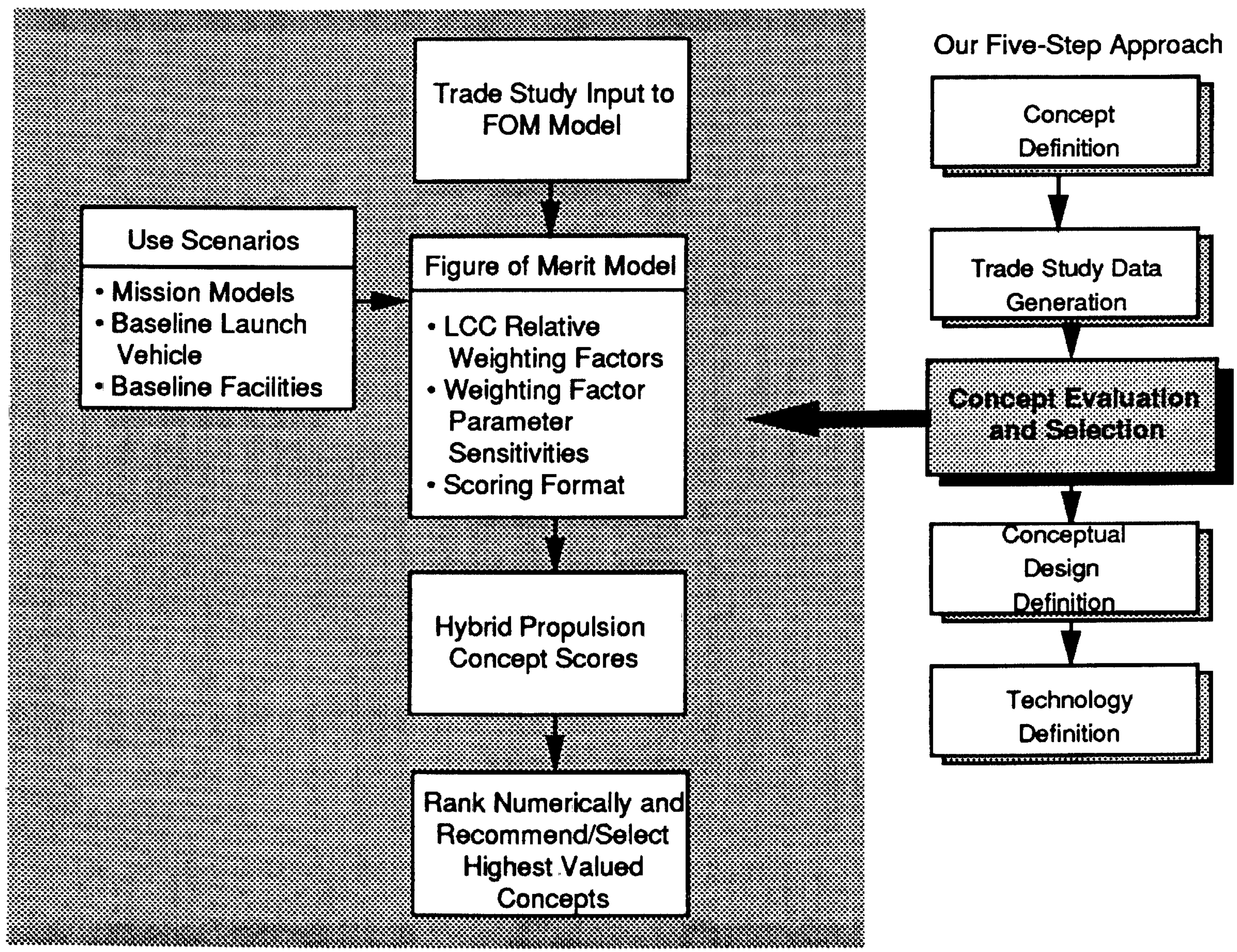
OUR CONCEPT RANKING METHODOLOGY IS QUANTITATIVE, USING COST AS FIGURE OF MERIT (FOM)



SELECT HIGHEST VALUED CONCEPTS

This Summary Chart Shows the Results of Eight HRB Design Concepts From the Ten Concept Selection Studies We Ran. The Large, Expendable HRB Flown Once Per Week Has the Highest Figure of Merit Score When It:

- Burns Propellant Number 8 With LO_2
- Use the Solid/Liquid Staged Combustion Concept
- Is Turbopump Fed With the Expander Bleed Burnoff Turbine Drive Cycle
- Uses Four Thrust Chamber Assemblies, Designed for Safe Abort With One TCA "Out", but Not to Make Mission With TCA Failure and Full Payload
- Uses Four LO_2 Turbopumps (One Per TCA)
- Uses One Solid Case
- Uses GO_2 Tank Pressurization for LO_2
- Uses a Cylindrical Solid Case With a Conical Aft Portion
- Uses a Cylindrical LO_2 Tank With an Integral Nose Cone (Like the ET)

These Results Do Not Vary With HRB Scenario, Except:

- Small HRBs Use One TCA and TPA (Unchanged at 8 Each/STS Flight)
- Small HRBs Have Cylindrical Solid Cases

We Also Ran Two Other Studies to Define Relative Scores for Large vs Small and Reusable vs Expendable HRBs

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CONCEPT SELECTION SUMMARY FOR EIGHT USE SCENARIOS

	Best Scores							
Scenarios	1	2	3	4	5	6	7	8
Reusable	No	No	No	No	Yes	Yes	Yes	Yes
No. HRB Flights	2	2	8	8	2	2	8	8
Flight Rate	1/wk	1/mo	1/wk	1/mo	1/wk	1/mo	1/wk	1/mo
Concept Selections								
Level 1 Propellants	LO_2 + #8	→	→	→	→	→	→	→
Level 2 Combustor	SLSC (D)	→	→	→	→	→	→	→
Level 3 Feed System	TF/EBBC	→	→	→	→	→	→	→
Nozzle Exit Pressure	41.37 kPa (6 psi)	→	→	→	→	→	→	→
No. TCAs, 0 "Out", HRB	4	4	1	1	4	• 4	1	1
No. TPAs/HRB	4	4	1	1	4	4	1	1
No. Solid Cases/HRB	1	1	1	1	1	1	1	1
Solid Case Shape and	Coni-Cyl	Coni-Cyl	Cyl	Cyl	Coni-Cyl	Coni-Cyl	Cyl	Cyl
Tank Shape	Cone/Cyl Rev. Hd.	→	→	→	→	→	→	→
TCA Cooling (Throat)	LO_2 Regen	→	→	→	→	→	→	→
Tank Pressure	Autog.*	→	→	→	→	→	→	→

*Turbine Exhaust Bleed - No Heat Exchanger or Regulator Required

FIGURE OF MERIT MODEL

Ed Bair Is Aerojet's Project Engineer for NASA/MSFCs ALS STME/BE Studies. We Used Several of the Concepts and Data in This Paper for This HRB Study.

Ed Selected a Bleed Cycle Liquid Engine for ALS With the Method and Data He Describes in This Technical Paper

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AIAA'88

AIAA-88-3213

Propulsion System Life Cycle Cost Screening - An Objective Selection Process

By:
E.K. Bair
Aerojet TechSystems Company
Sacramento, California

AIAA/ASME/SAE/ASEE
24th Joint Propulsion Conference and Exhibit

July 11 - 13 1988
Boston, Massachusetts

It Was Found That Life Cycle Cost Can Be Described Adequately in Five Segments, and That Their Relative Importance Changes With the Number of Missions Made. We Used This Same Data for Our HRB Study. Notice that the Individual Weighting Factors in the Graph Add up to 100 Points (Percent) for All Mission Numbers

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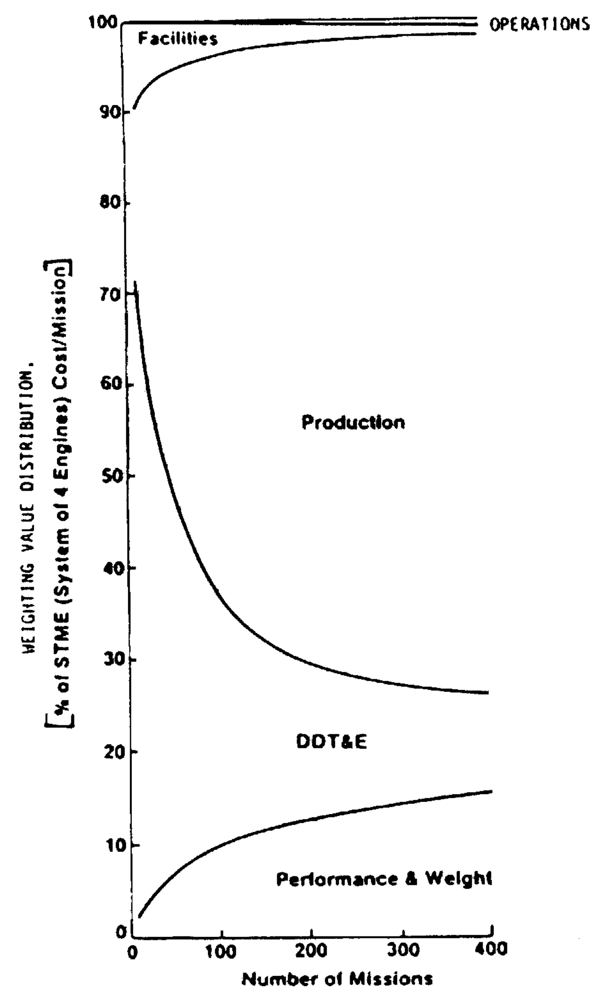
CATEGORY COST AND WEIGHTING

Category	10 Missions		100 Missions		400 Missions	
	Value/Mission	Weighting Value	Value/Mission	Weighting Value	Value/Mission	Weighting Value
Performance and Weight	5.55×10^6	2.5%	5.55×10^6	10.0%	5.55×10^6	15.5%
Development	150×10^6	67.8%	15×10^6	27.1%	3.75×10^6	10.5%
Production	45.56×10^6	20.6%	32.6×10^6	59.0%	25.8×10^6	72.2%
Facilities	20×10^6	9.0%	2×10^6	3.65%	$.5 x 10%	1.4%
Operations	$.14 \times 10^6$	.1%	$.14 \times 10^6$	.25%	$.14 \times 10^6$	.4%
Total	221.25×10^6	100%	55.29×10^6	100%	35.72×10^6	100%



This Chart Defines Our HRB Study Life Cycle Cost Element Weighting Factors. It Was Created From the Previous Chart by Inputting the Two Total Numbers of Flights From the Scenario Table on Page 138. These Become the Maximum "Scores" That HRB Candidates Can Earn in the Five LCC Categories

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WEIGHTING FACTORS FOR LCC EVALUATION—HRB STUDY

	Life Cycle Cost Elements				
STS Flight Rate	Payload	Development	Production	Facility	Operations
1/mo.	29.0	16.5	51.9	2.4	0.2
1/week	38.0	5.2	55.6	0.8	0.4

Actual Scores Earned by HRB Candidates Depended Upon Their Values for LCC - Sensitive Parameters as Well as the Maximum Score. The Following Two Pages Define the Algorithims Used to Determine Actual Score Elements. Payload Score Depends on Payload to LEO Capability. Less Than ASRMs 29.5 Metric Tons (65 klbm) Capability Gives a Negative Score. We Did Not Evaluate Payloads Greater Than 40.8 Metric Tons (90 klbm) With the Current STS. Production and Development Cost Scores Depend Upon AUPC Per Flight Set and HRB Development Cost Totals, Respectively

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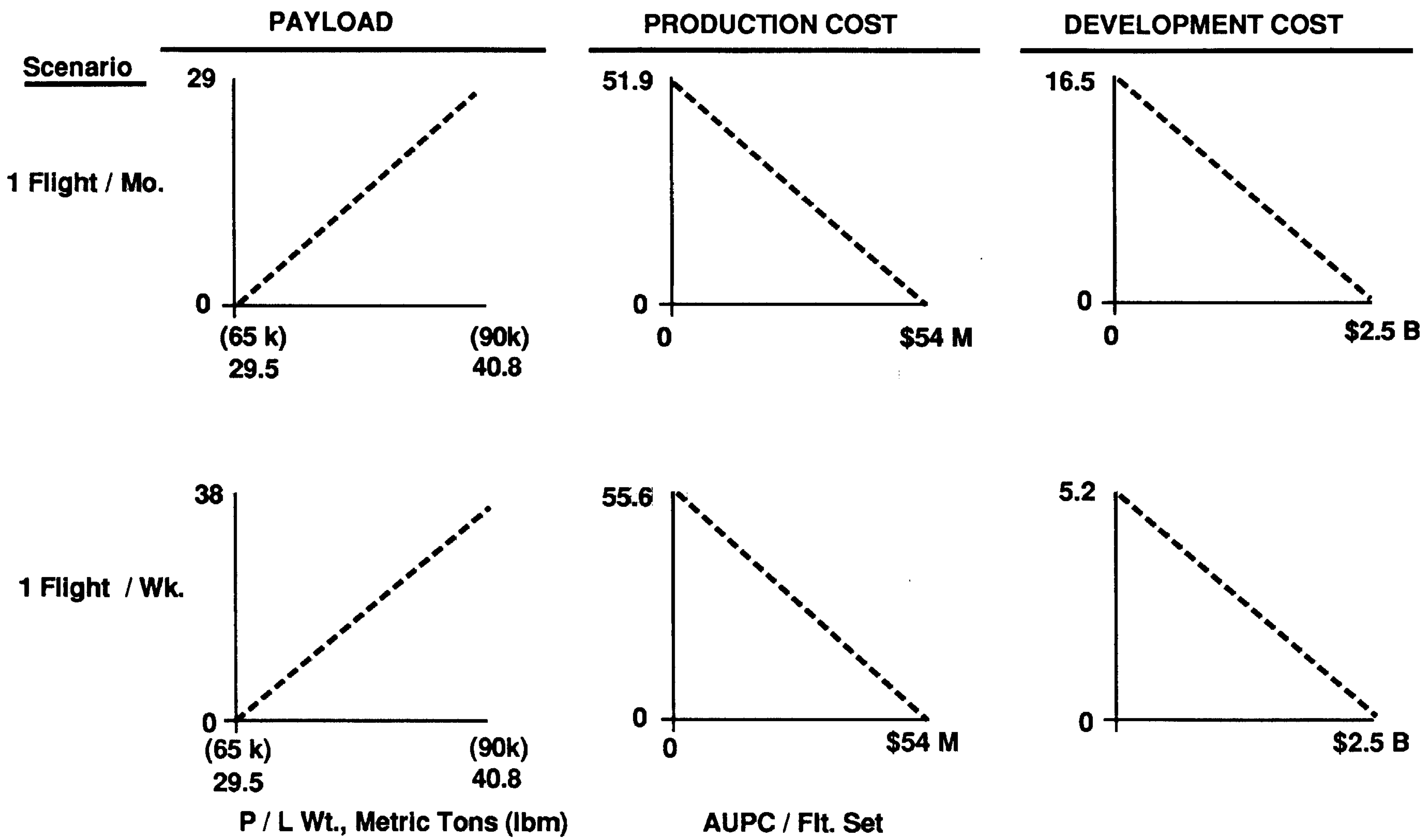
WEIGHTING SENSITIVITY FACTORS, SHEET 1 OF 2



3.16.0.13

Similarly, Facilities and Operations Scores Depended Upon Facility and Operation Cost Differences Between Candidates. Shaded Portions of the Graphs Defined the HRB Study Ranges

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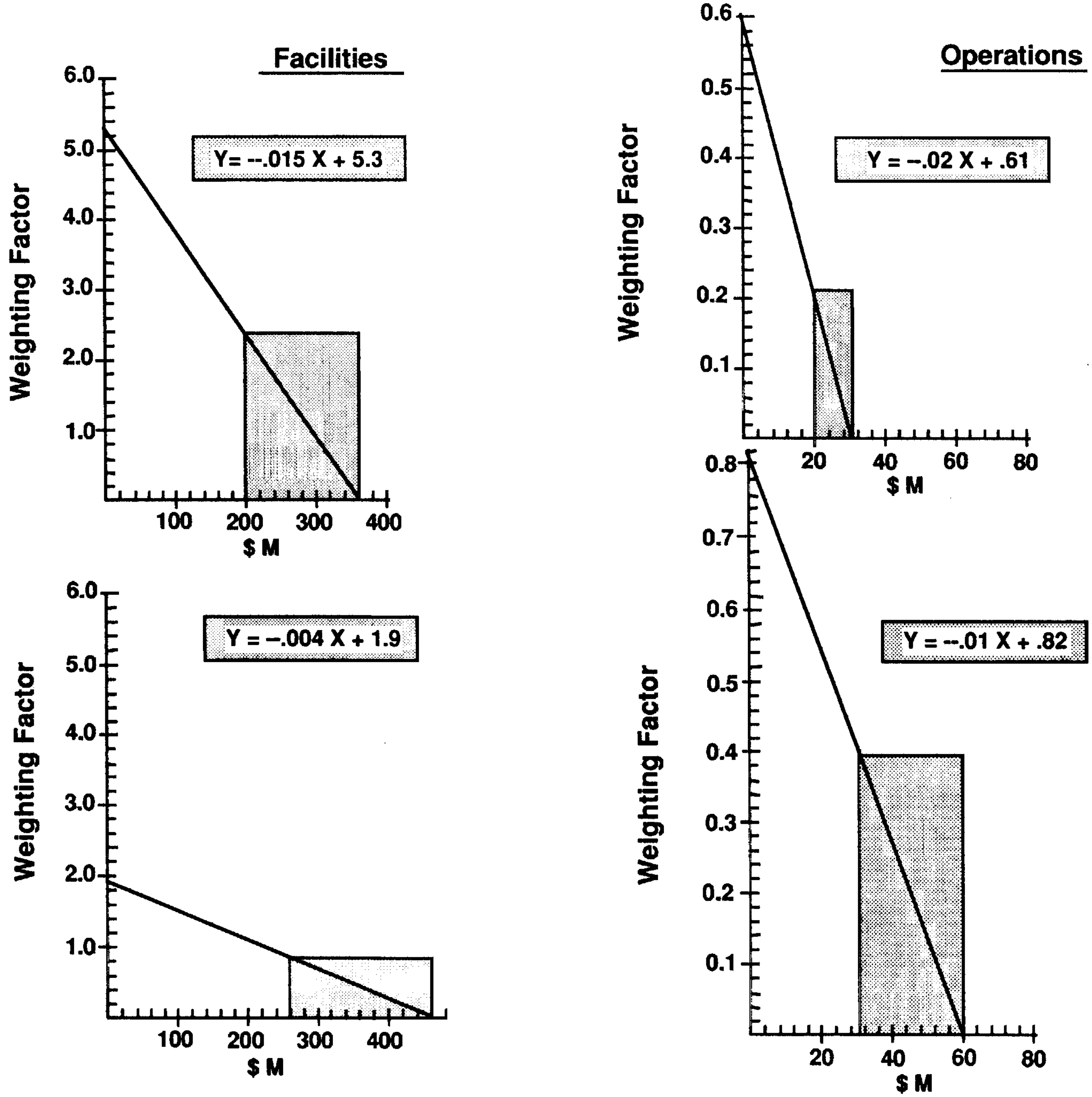
WEIGHTING SENSITIVITY FACTORS, SHEET 2 OF 2

Scenario

1 / Month

1 / Week



3.16.0.14

This Table Summarizes the Scope of Each LCC Element, Defines the Differences in Each Category Needed to Generate One Point of Score Difference for the Two Flight Rate Scenarios. It Also Shows the Values of 1 kg (1 lb) of Payload Capability to LEO in Terms of the Other Four LCC Elements

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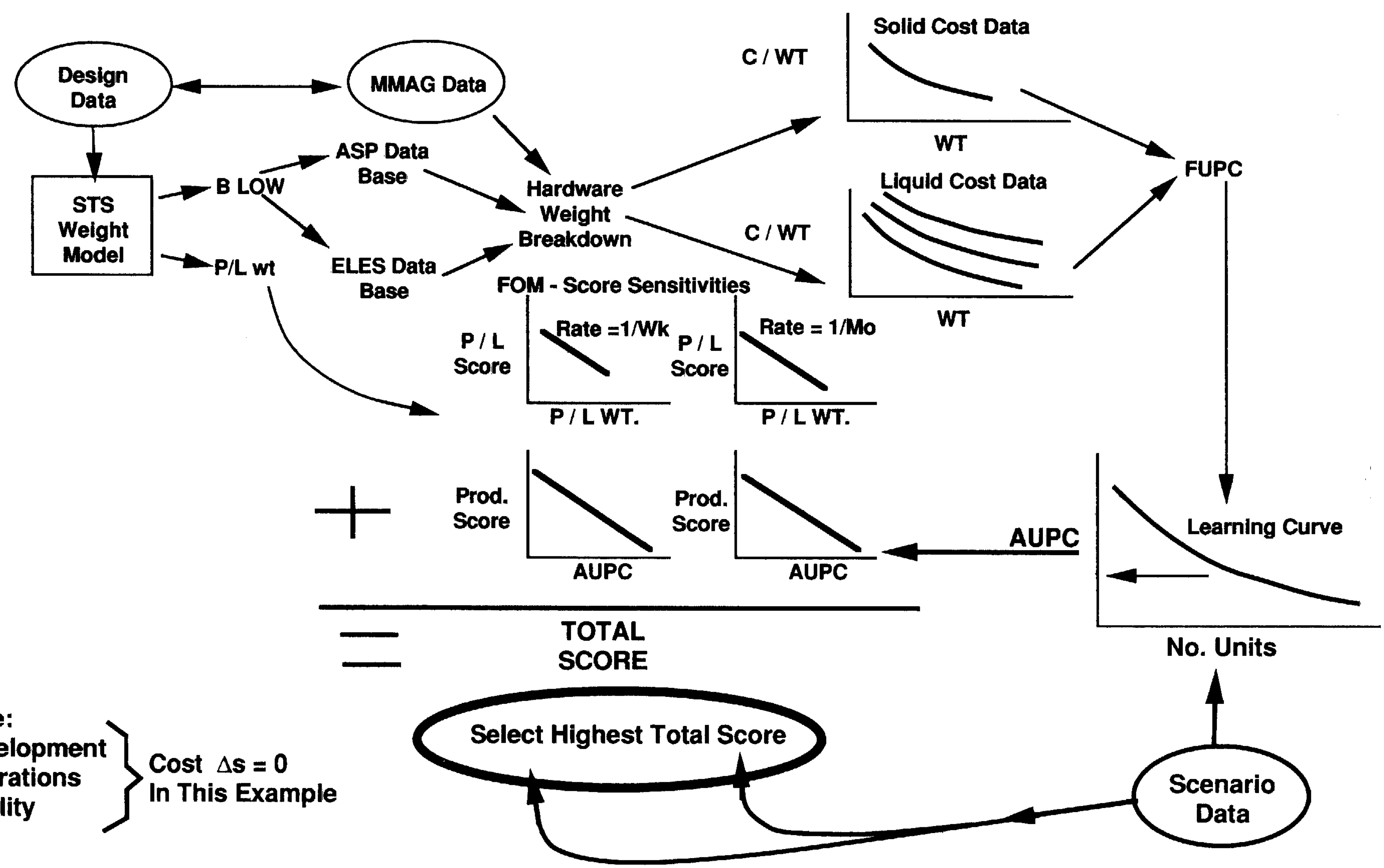
ONE POINT IS SCORED BY ANY OF THESE CANDIDATE CONSIDERATIONS

Scenario	Payload	Production Cost	Development Cost	Facility	Operations
1 Flt/mo	391 kg (862 lbm)	$520K	$120M	$67.9M	$49.2M
	(1) ⟶	$1,329/kg ($603/lbm)	$306K/kg ($139K/lbm)	$173.7K/kg ($78.8K/lbm)	$125.9K/kg ($57.1K/lbm)
1 Flt/wk	298.5 kg (658 lbm)	$485K	$385M	$242M	$73.2M
	(1) ⟶	$1,625/kg ($737/lbm)	$1,290K/kg ($585K/lbm)	$811K/kg ($368K/lbm)	$245K/kg ($111K/lbm)
Scope	Payload Delivered to LEO by Orbiter	Average Cost for HRB Flight Set. Includes All Refurbishment Costs, if Applicable	Engineering Manpower, Hardware, and Testing Costs. Cost of Any New Development Facilities Not Included	All Assembly, Launch, Logistics, Transporters, Workshops and Propellant Area Facilities	Labor Cost for Hardware Installation, Checkout and Test. Facility Maintenance Not Included

This Chart Illustrates the Numerical Method We Used to Calculate HRB Candidate Selection Scores. We Began With Our STS Weight Model Output (Booster Liftoff Weight and Payload to LEO). We Detailed the HRB Weights With Our Solid and Liquid Data Bases and Our Baselined HRB Weights With Our Propulsion Scaling Code, ELES. Next, Detailed Weights Were Converted to Detailed First Unit Production Costs With Our Cost/lbm Data Bases. Summed Costs Then Are Converted to Average Unit Production Costs With Learning Curve Analyses Driven by NASAs Use Scenario Data. Note That Other NASA Requirements Were Used as Input to Our Weight Model or In Candidate Screening in Task 1. The AUPC and Payload Data, in This Example, Are Input to the Scoring Algorithms of the FOM Model. Adding the Scores of Each Element of the FOM (in This Case Two Elements - Production Cost and Payload Weight Benefit) Yields the Final Candidate Score

EXAMPLE OF SCORING AND SELECTION METHODOLOGY



Note:
Development
Operations
Facility } Cost $\Delta s = 0$ In This Example

Using Our Weight Model as Input and the Method Just Described, We Performed Nine LCC-Based FOM Studies as Shown Below

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WE MADE NINE CONCEPT SELECTION STUDIES WITH OUR FOM MODEL*

Level 1	• 10 + 2 Propellant Combinations With TF and PF Vehicles With 2 Combustion Schemes
Level 2	• A, B, D, E Combustion Schemes With B/L Vehicle, Using No. 8 and 8b Propellant Combinations
Level 3	• Propellant Feed Cycle: PF vs TF - Bleed, TF - Topping, TF - BBC
	• Nozzle Exit Size 41.4 kPa or 3.78 m (6 psia or 149 in.) OD. Current MLP Geometry Constraint
	• One Engine Out vs No Engine Out Philosophy to Make Mission vs Number of Engines for Each, Using D Type TF Large HRB
	• No. of TPAs (Large HRB Only)
	• No. of Solid Motor Cases for Large HRB
	• Reusable vs. Expendable HRBs
	• Small vs. Large HRB

*For High and Low Flight Rate Scenarios, Plus Three Other Trade Studies

The Following Bar-Graph Charts Show the Results and Details of These Studies

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HRB CONCEPT SELECTION SCORES

- Level 1 — Propellant
- Level 2 — Combustor
- Level 3 — Subsystems

This Chart Shows the Solid Propellant Costs Used in Our Study. They Include Material Cost Estimates and a Small Multiplier for Mixing and Casting Into a Large HRB Based Upon Weight Model Results. Refer to Page 42 for Solid Propellant Compositions. Blends 7, 12, and 13 Contain Boron, an Expensive Ingredient. Blend 9 Has a Significant Amount of BAMO Content, a Currently Expensive Ingredient

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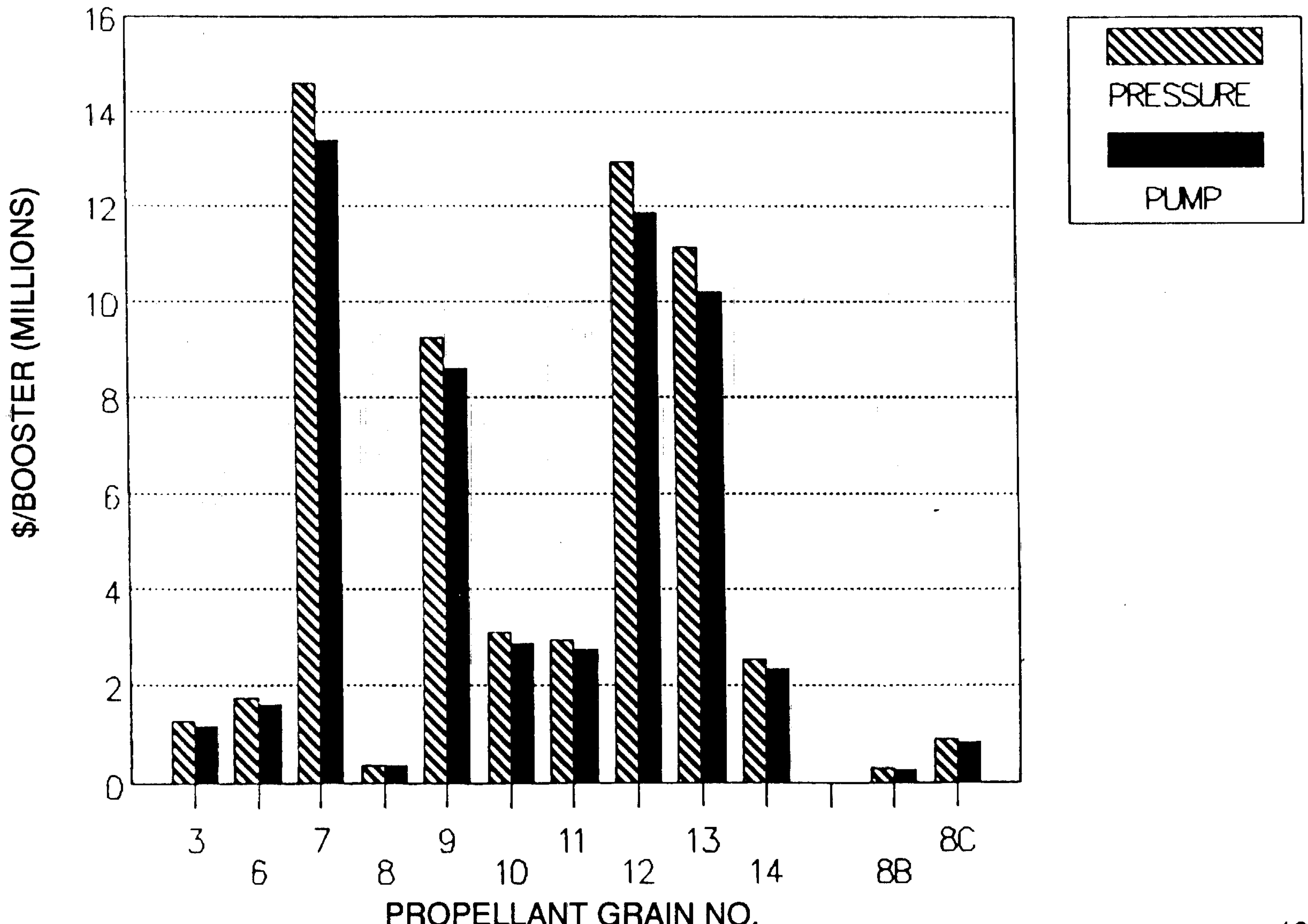
HRB SOLID PROPELLANT COST



Based on the Foregoing Data, HRB Scores With Various Propellant Combinations Were Calculated and Are Shown Below. Left Hand Data Is for Fuel-Rich Solid Propellant, While Right Hand Data Is for Fuel-Only Solid Propellant. By Changing the Propellant Data Only (Not I_{sp} Efficiency or Any Other Parameters) It Is Clear That Propellants 8B and 8 Should be Selected, on the Bases of Higher Scores. Propellant Combination Number 9 Was Screened Out, Because It Uses (Unsafe) H_2O_2, and This Chart Shows That It Provides a Potential Improvement of Only About 1%

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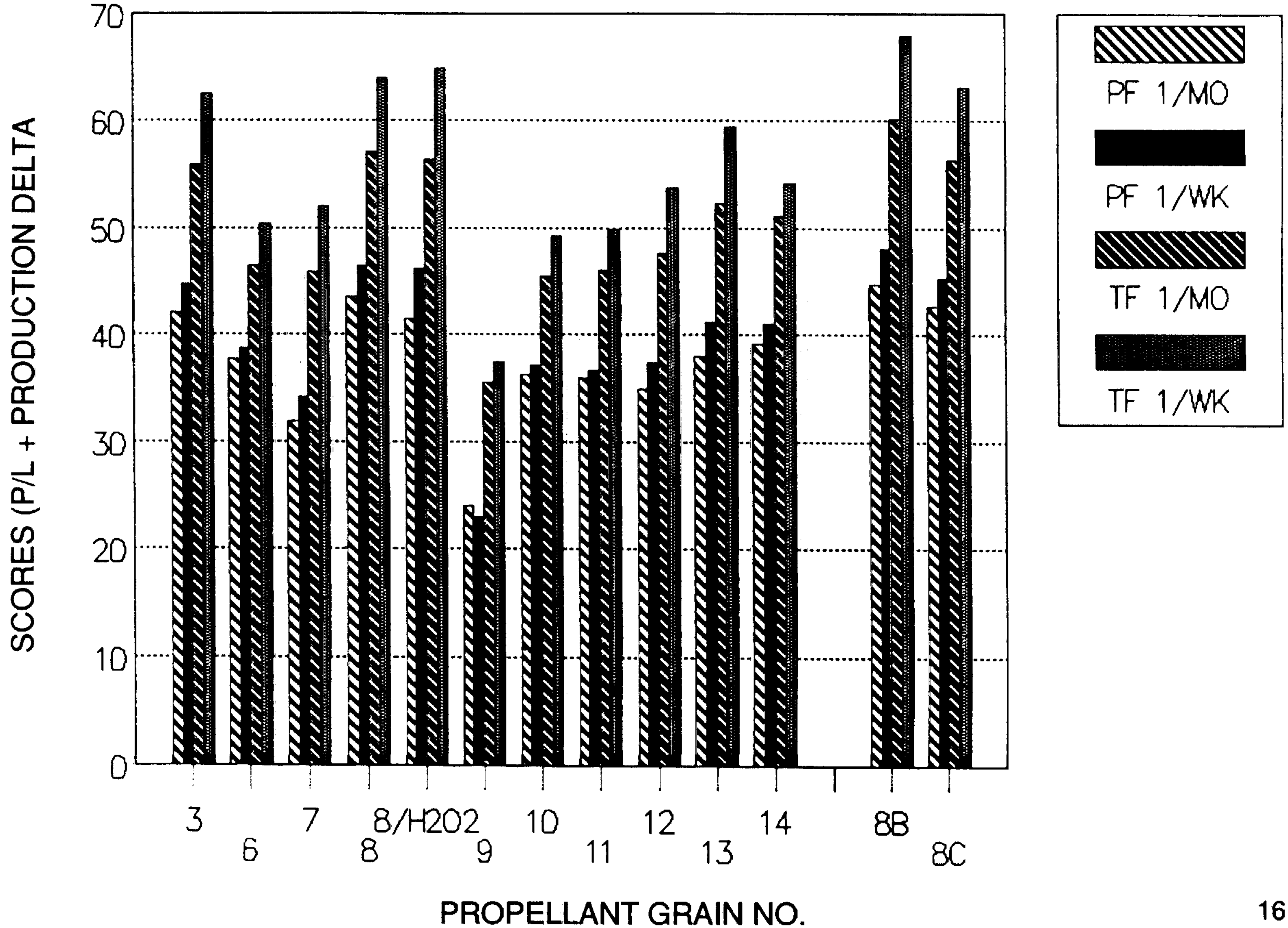
HRB SOLID PROPELLANT CANDIDATES



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Solid Propulsion

HRB EXHAUST CONSTITUENTS*

Engine Exhaust Products		Atmosphere Burnoff Products	
Compound	Percentage	Compound	Percentage
H_2O	46.3	H_2O	25.8
CO_2	36.5	CO	16.5
CO	9.8	CO_2	15.8
H_2	2.3	N_2	15.4
N_2	1.7	O_2	7.5
NaCl	1.2	OH	6.0
Other	2.2	H_2	3.8
		H	2.9
		O	2.8
		NO	1.1
		Other	2.4

*With LO_2/PEBC and Tufflo Scavenger, Propellant Number 8

This Chart Shows Why We Selected the Solid-Liquid Staged Combustion Concept Over Three Other Candidates. Conventional and Quasi-Hybrids Had Lower Scores, Because Their Payload Carrying Capability Was Poor and Their Liftoff Weight High. Both Are Caused by Low Specific Impulse, Derived From the Poor Mixing Efficiency Legacy of Forward-Injected, Single Stage Combustion Hybrid Concepts. Their Production Costs Are High, Because of Their High Weight and Failure to Have Simpler Hardware Than Aft Injected Concepts. The Fourth Candidate, Hybrid Solid Case Gas Generator Type SLSC Has Two Injectors and a Slight Performance Decrement Due to Nonuniform Hybrid Fuel-Rich Gas Preparation. Thus, Both Its Payload and Production Scores Are Less Than the SLSC

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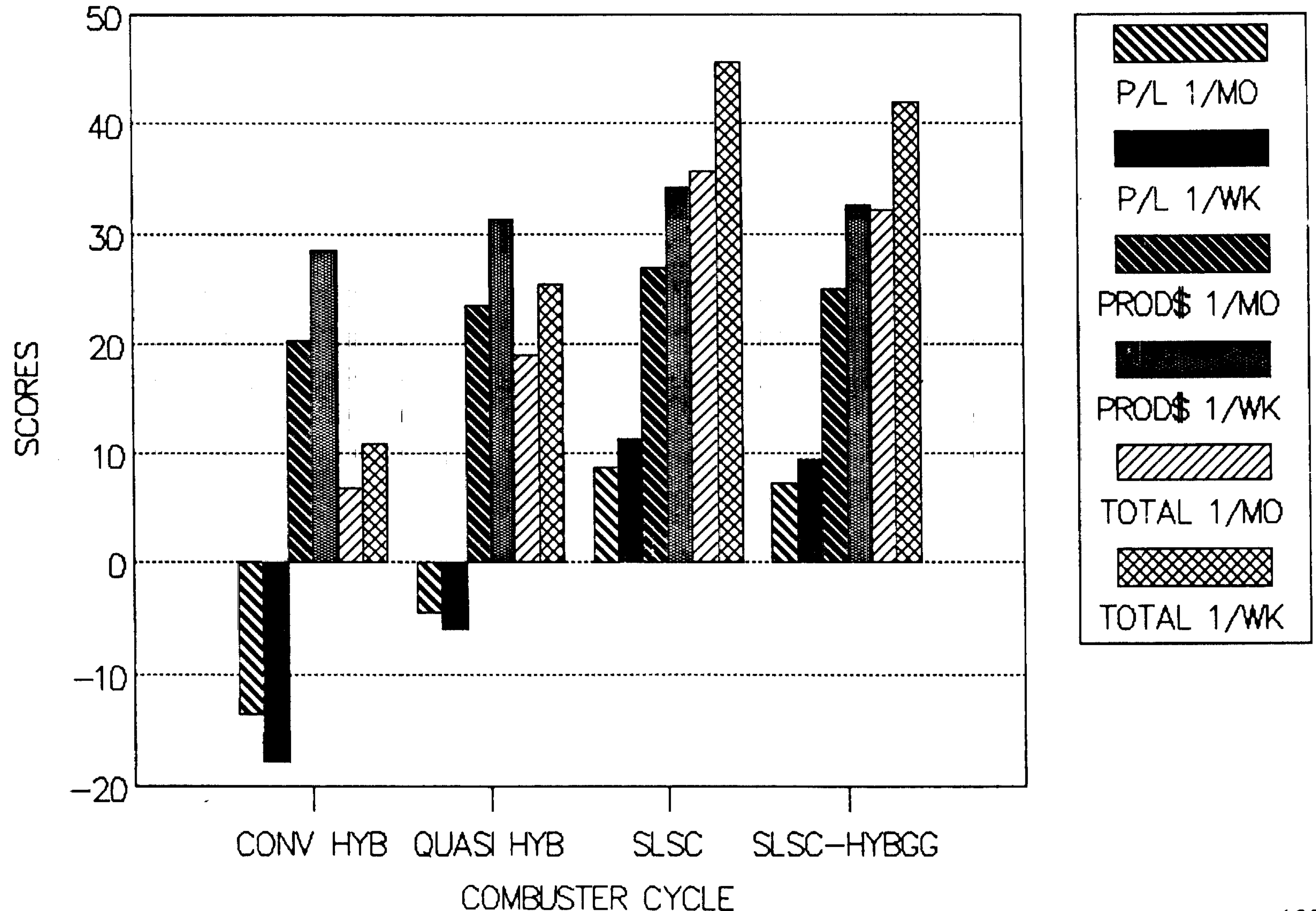
HRB COMBUSTOR CONCEPT SELECTION



Turbopump Fed HRBs Scored Much Better Than Pressure Fed Designs When the Turbine Is Driven With Bleed Gases (Not by Gases to Pass Through the Injector). Pressure Fed Variants Suffered From Low Payload Delivery to LEO, Because of Heavy Tankage and Pressurization Weights. The Topping Cycle Score Was Lower Than the Bleed Cycle, Because Its Low Specific Impulse Hurt Its Payload Capability. The Low I_{sp} Is Caused by the Relatively Poor Combustion Efficiency of a Gas/Gas Injector When Used With O_2/Hydrocarbon Systems

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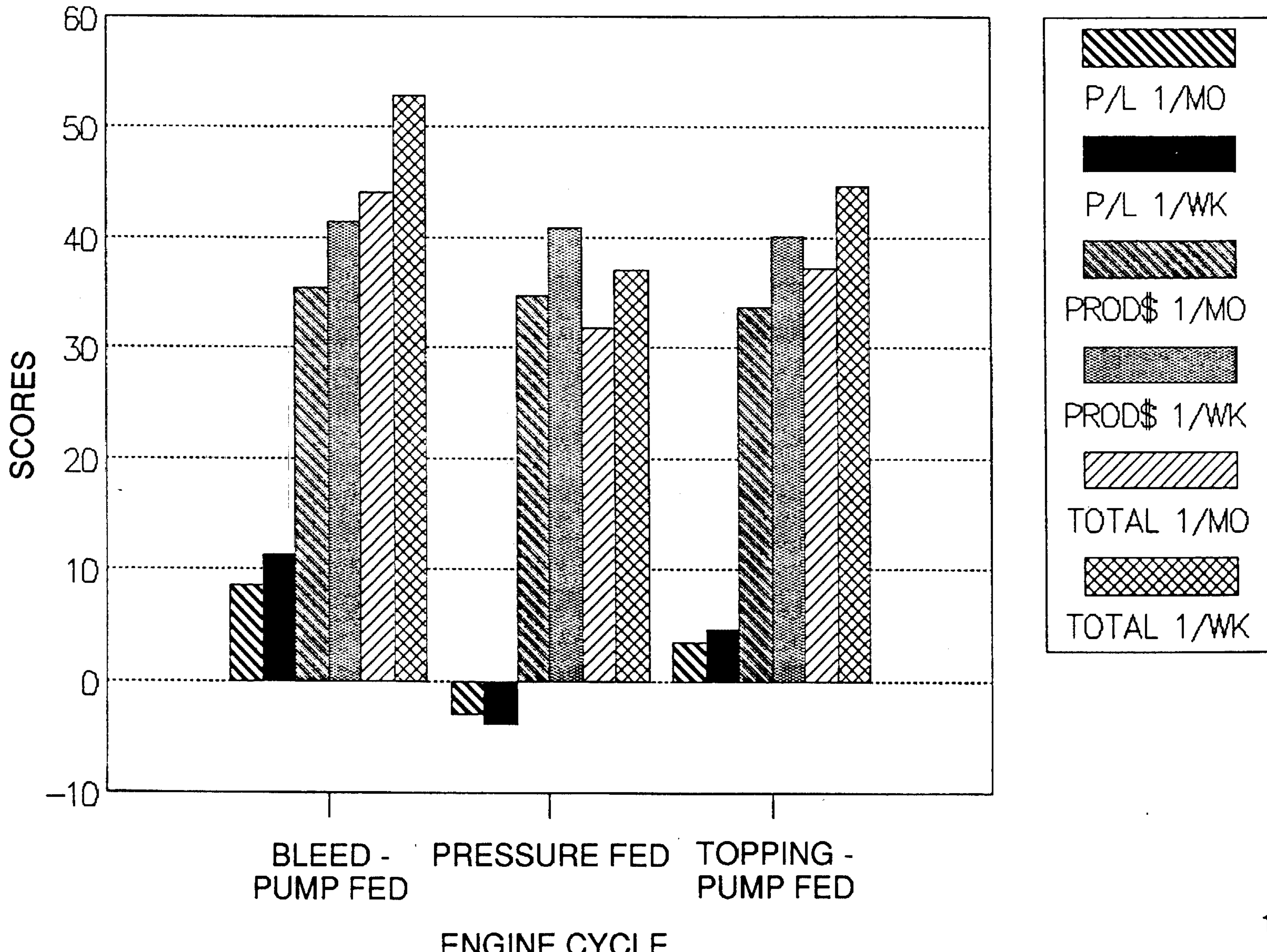
HRB LO_2 FEED CYCLE SELECTION



We Selected Large Area Ratio Nozzles Because They Scored Better Than Small Nozzles, Depending Upon the Design and Use Scenario. For Our Selected Design, Turbopump Fed Large Nozzle Advantage Can Be as Large as 15 to 20%. More Frequent Flights Accentuate the Payload Carrying Advantage of Large Nozzled Boosters. The Smaller TF and PF Nozzles Fit the Current Mobil Launch Platform (for Single Nozzled HRBs), and the Larger Ones Expand to the Generally Accepted "Best" Value of 41.36 kPa (6 psia)

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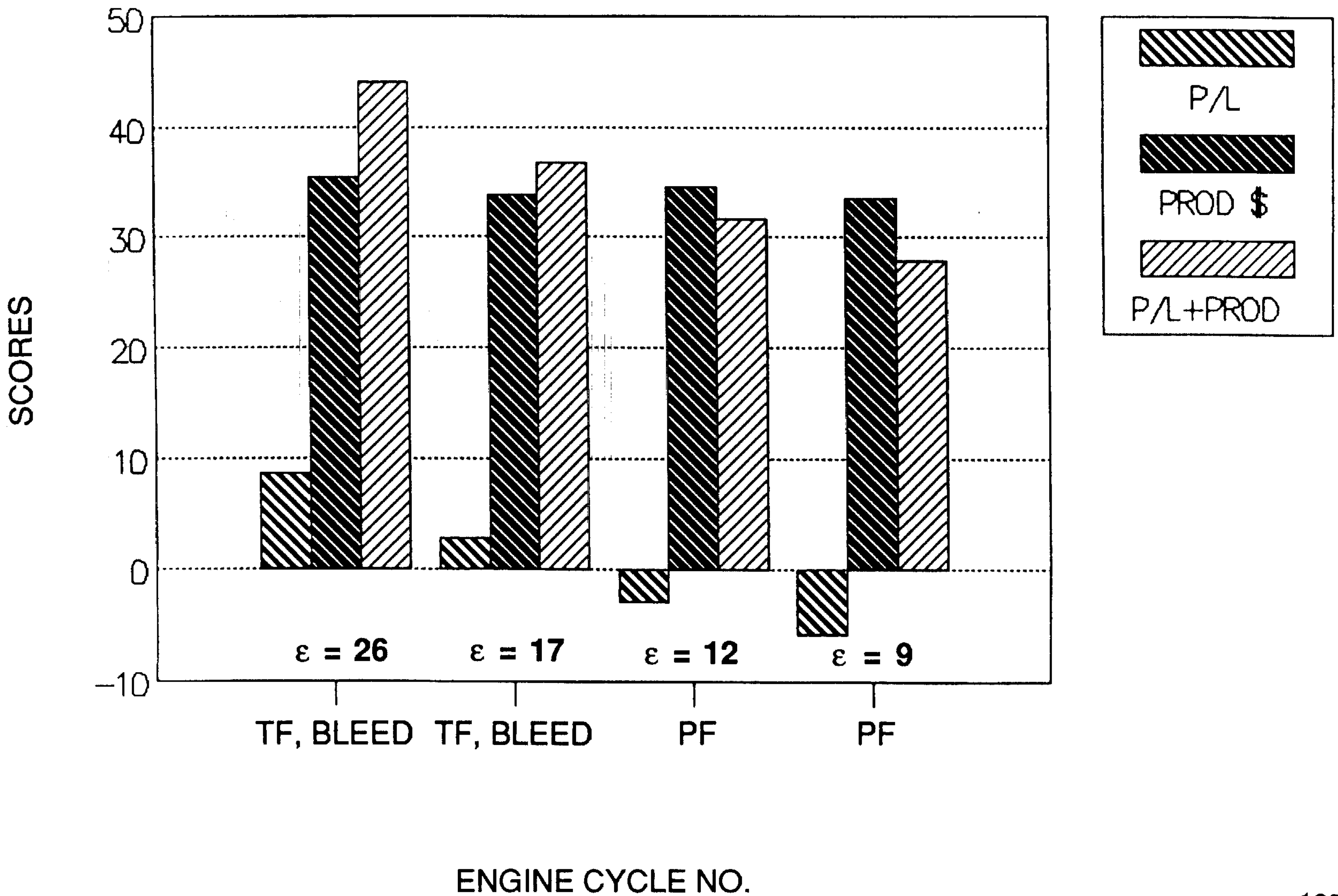
EXIT PRESSURE/NOZZLE GEOMETRY SUBSYSTEM OPTIMIZATION SELECTION



In Order to Assess the Proper Number of Thrust Chambers to Use for Large HRBs, It Was Necessary to Define Our Goal; i.e., to First Define Our Engine Reliability Philosophy

Fixing Propulsion Reliability and Backing Out Engine Reliability vs Number of Engines Yields Unacceptable Development Costs in Most Cases. Thus, the Opposite Approach Was Used—Fix Engine Reliability and Determine the Cost of Unreliability. However, There Are Two Different Results, Depending Upon Whether We Design to Make Mission or Have a Safe Abort With One Engine Out on a Large HRB

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HRB RELIABILITY PHILOSOPHY ASSESSMENT

Select:

- Number of Engines/Large HRB, Based on:
  - — Mission Results for One TCA Failure

Method:

- Fix Engine Reliability Requirement
  - — Determine Unreliability Costs
  - — Fixes Engine Development Costs
  - — Assume No Category 1 Failures
  - — Therefore Failed TCA Mission Result Choices Are:
    - Make Mission
    - Safe Abort

This Table Shows the Number of Missions Aborted (vs Flight Rate Scenario) for 99% Reliable Engines vs Number of Engines Per Large HRB, With Single Engine "Out" Capability to Make Mission, vs With No Such Capability—in Other Words, to Abort Safely When an HRB Engine Fails During Launch

Reliability Was Calculated Using a 50% Fratricide Factor—Larger Than Used for Aircraft

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COST OF UNRELIABILITY

# ENGINES	SINGLE ENGINE RELIABILITY	SYSTEM RELIABILITY	# ABORTED MISSIONS	
			1/MO	1/WK
WITH SINGLE ENGINE OUT CAPABILITY				
1	.99000	.990000	1.44	6.24
2	.99000	.998517	.21	.93
3	.99000	.990788	1.33	5.75
4	.99000	.975668	3.50	15.18
5	.99000	.954874	6.50	28.16
WITH NO ENGINE OUT CAPABILITY				
1	.99000	.990000	1.44	6.24
2	.99000	.980100	4.28	12.42
3	.99000	.970299	4.28	18.53
4	.99000	.960596	5.67	24.59
5	.99000	.950990	7.06	30.58

Another Part of the Problem Is the Delivered Specific Impulse vs Number (and, Thus, Thrust Level and Size) of Engines. This Chart Shows That Engine Performance Will Drop With Increasing Engine Thrust Levels, Based Upon a Propellant Vaporization Rate Combustion Stability Parameter

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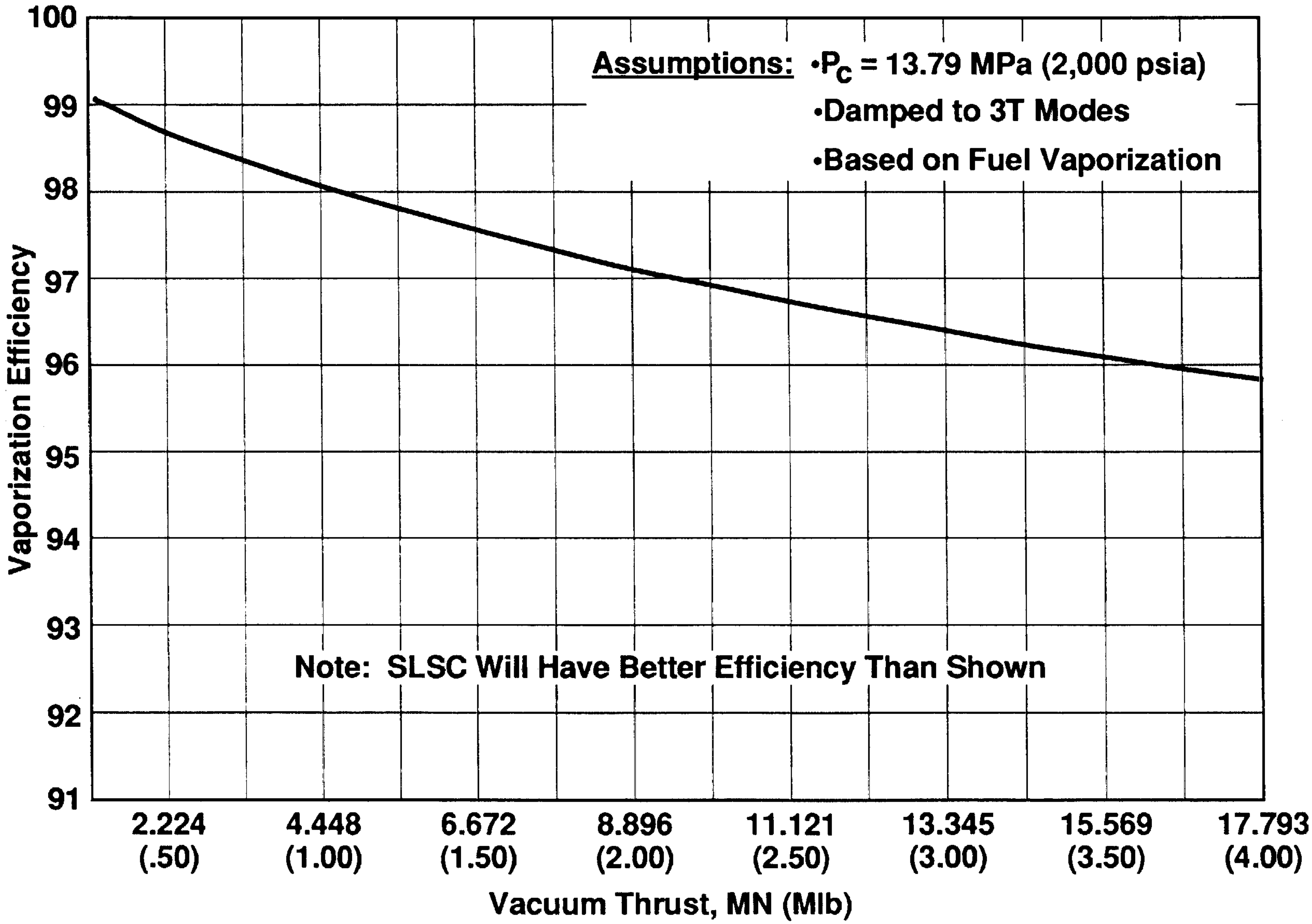
MAXIMUM SPECIFIC IMPULSE DEPENDS UPON ENGINE THRUST LEVEL



This Chart Shows Scoring Results for Two Through Five Engines Per Large HRB When Designed to Make Mission With One Failed Engine. It Shows That Five Engines Have the Best Score; However, Only About 3% Better Than Four Engines. Noting That Three, Four, and Seven Engines Package Well at the Base of a Round Vehicle, Four Engines Is a Good Choice for a Large HRB Design

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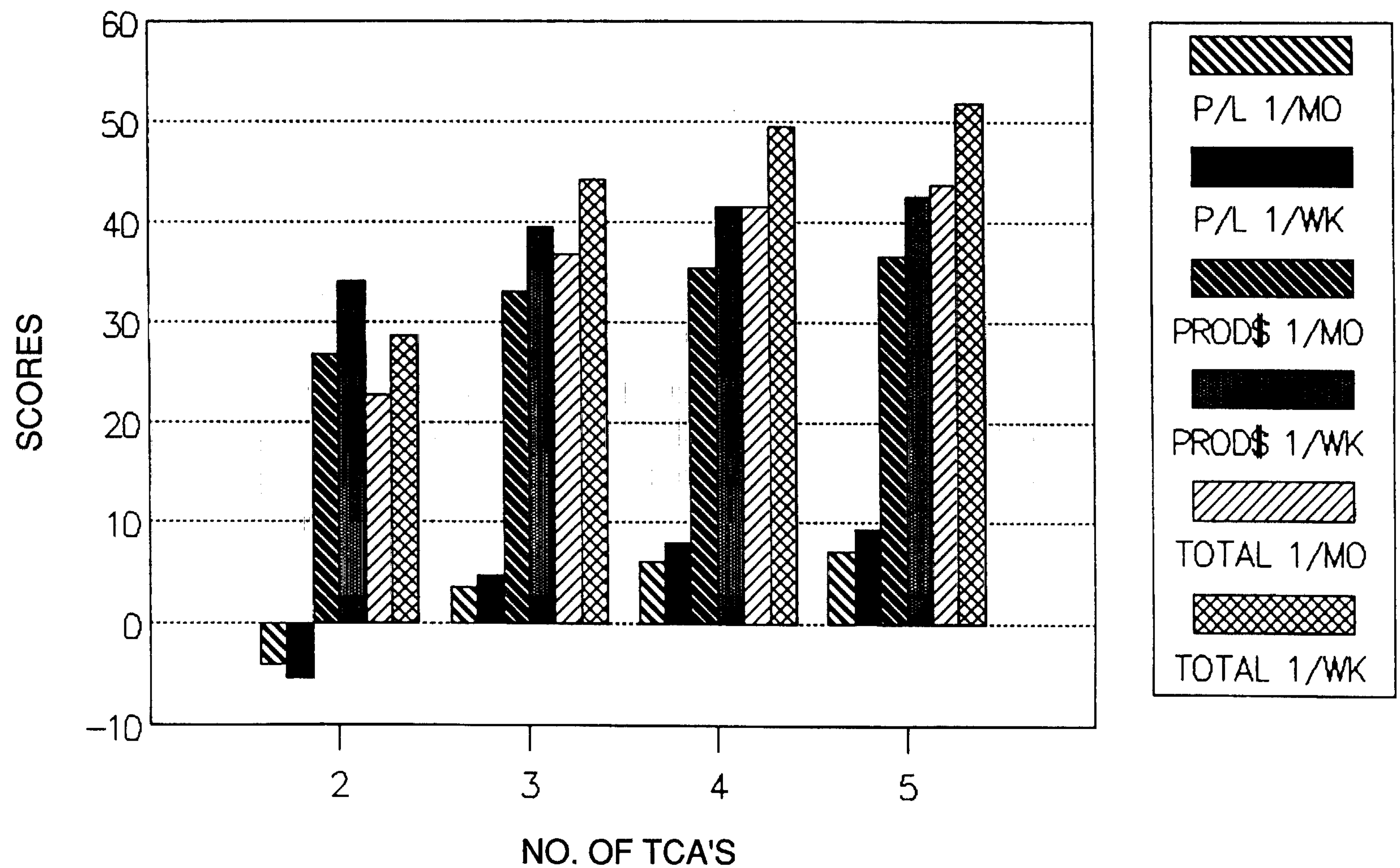
TCA SELECTION
SINGLE TCA OUT TURBOPUMP FED



This Chart Shows Similar Results When Engines Are Designed for a Safe Abort Only With a Failed Engine; Four Engines Are Preferred. Note the Higher Scores Here by About 3 or 4%. This Suggests That Less Expensive Engines and More Aborts Is Superior to the Opposite Design Philosophy, Even for a Large Number of Engines

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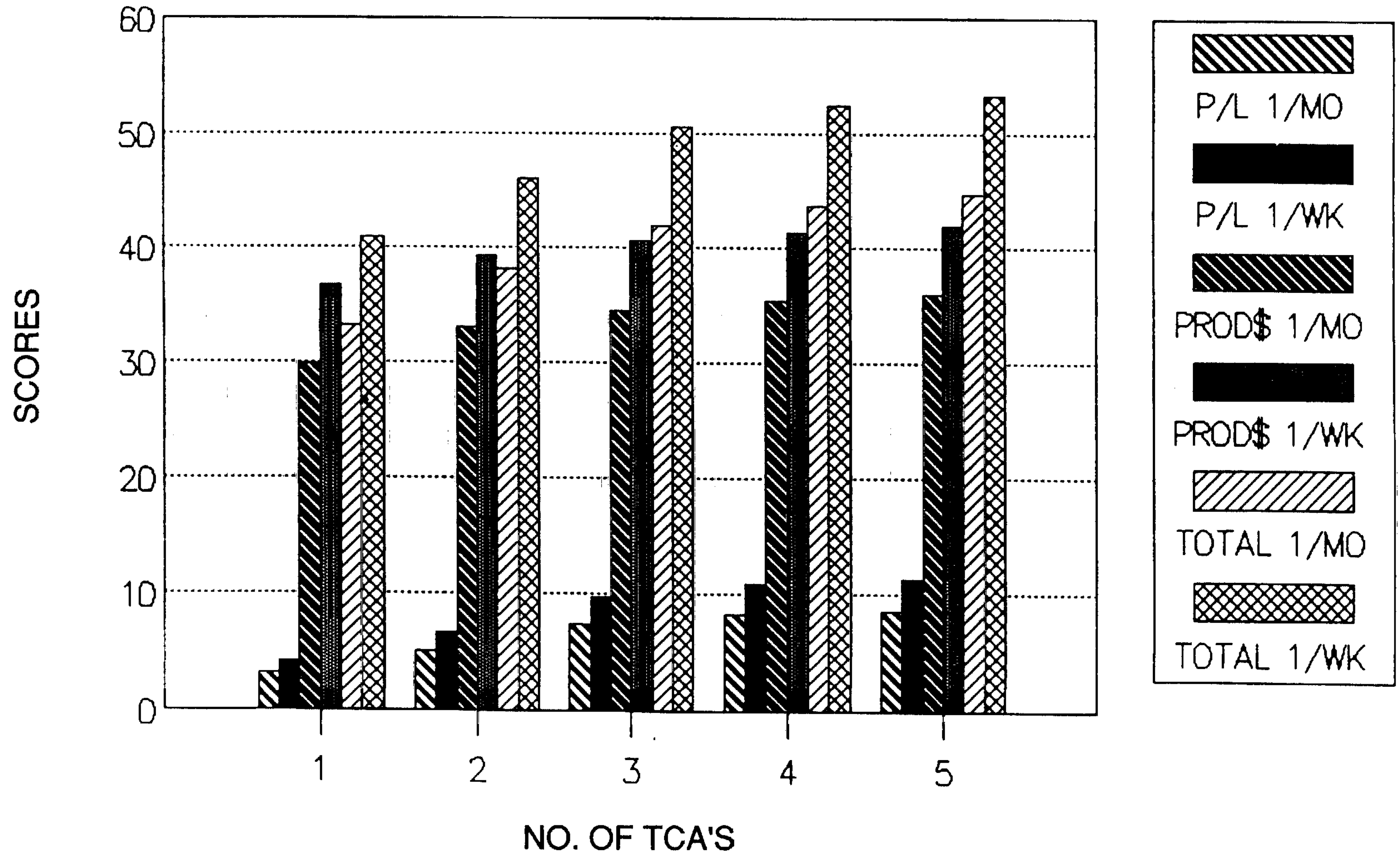
TCA SELECTION
NO TCA OUT TURBOPUMP FED

C-4



The Next Four Charts Show the Cost Data That Contribute to the Previous Score Charts, Covering the Four Flightrate/Engine Failure Philosophy Scenarios. Note That in Every Case the Total Cost Data Has a Minimum (vs. Number of Engines) Caused by Opposing Effects of Decreased Production Cost and Increased Unreliability Costs as Numbers of Engines/HRBs Increases

The Payload Effect of Changing Engine Specific Impulse Is Not Included, as in the Foregoing Score Charts

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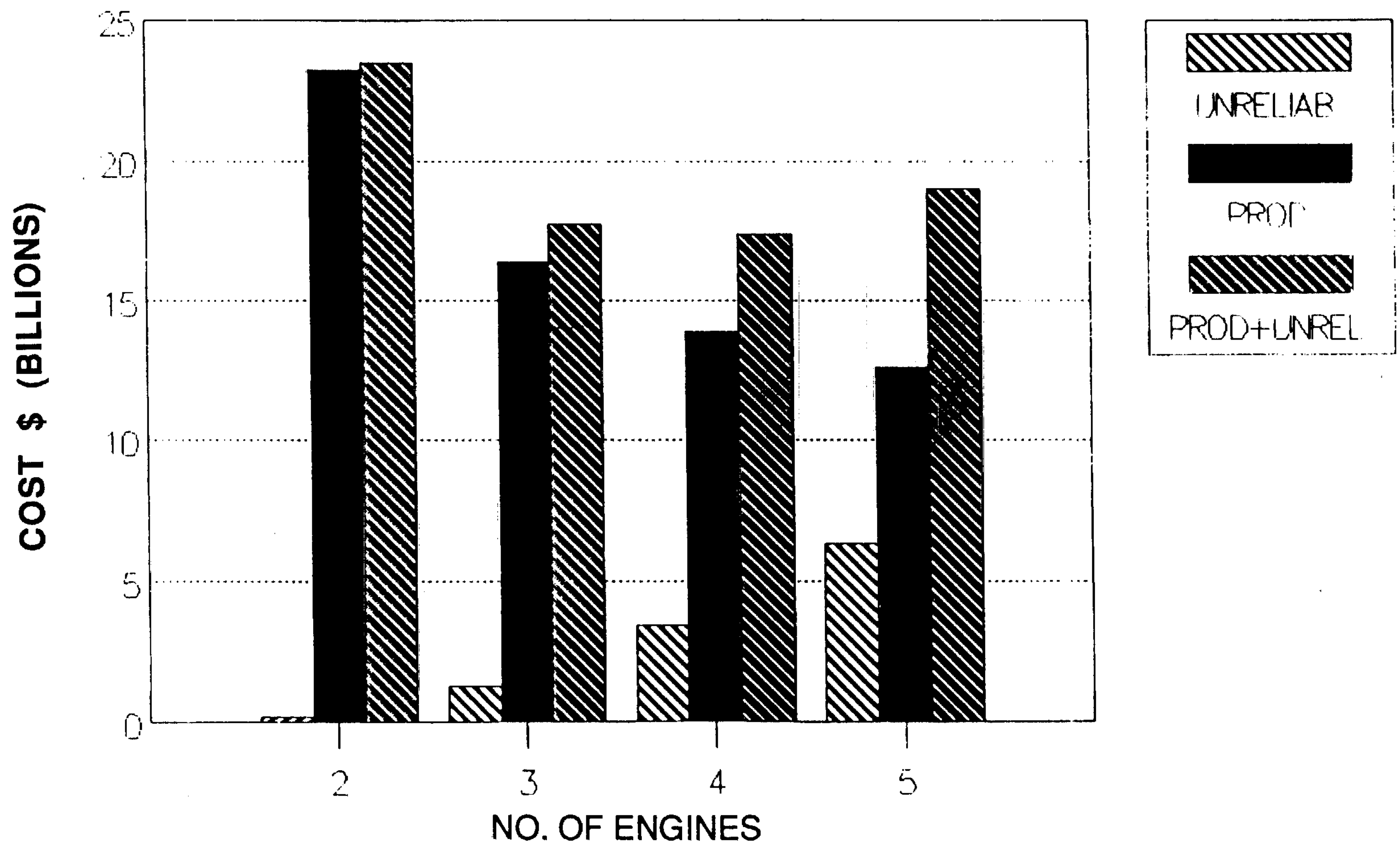
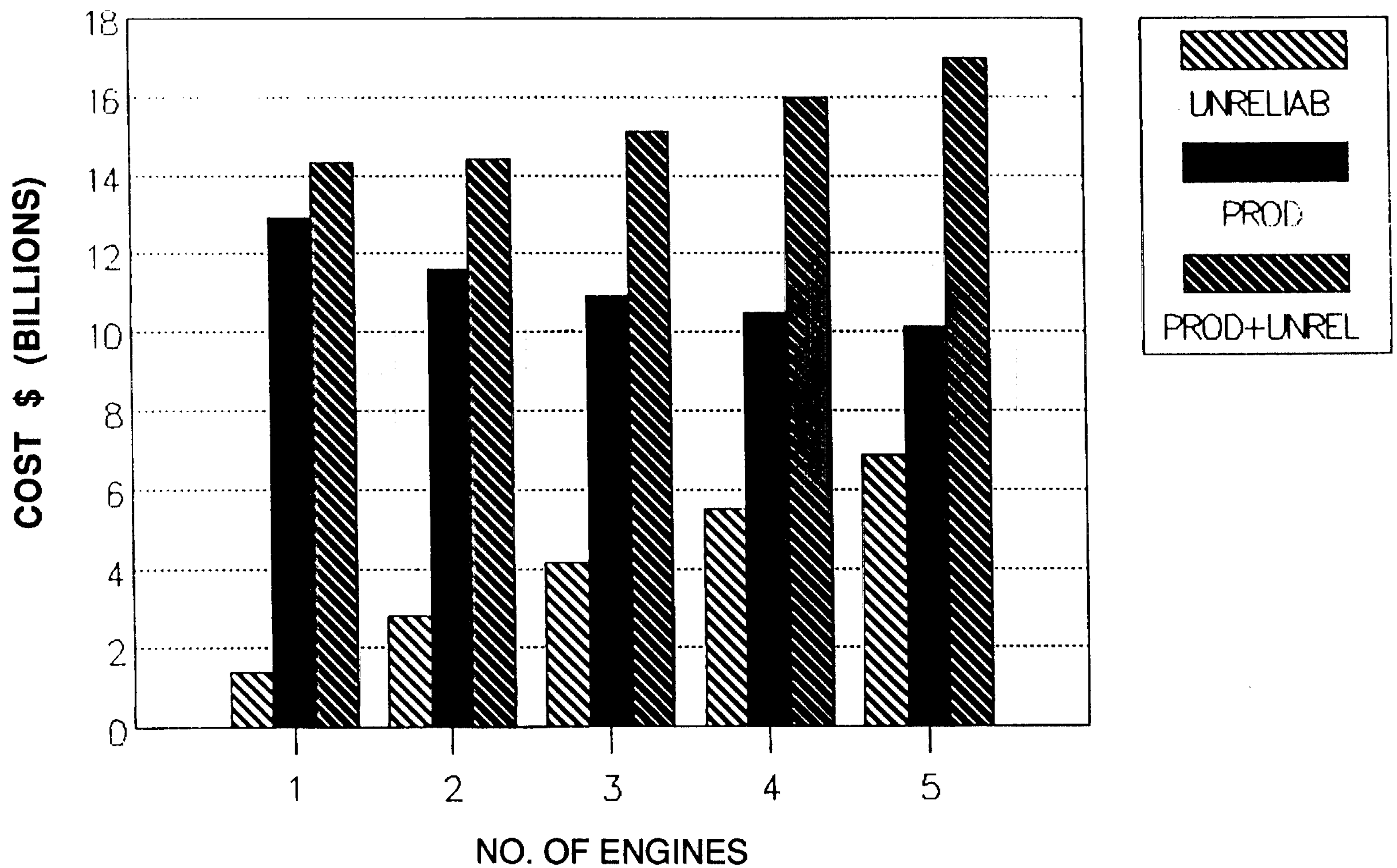
HRB ENGINE RELIABILITY TRADES SINGLE ENGINE OUT 1/WK LG EXPENDABLE



HRB ENGINE RELIABILITY TRADES
NO ENGINE OUT 1/WK LG EXPENDABLE



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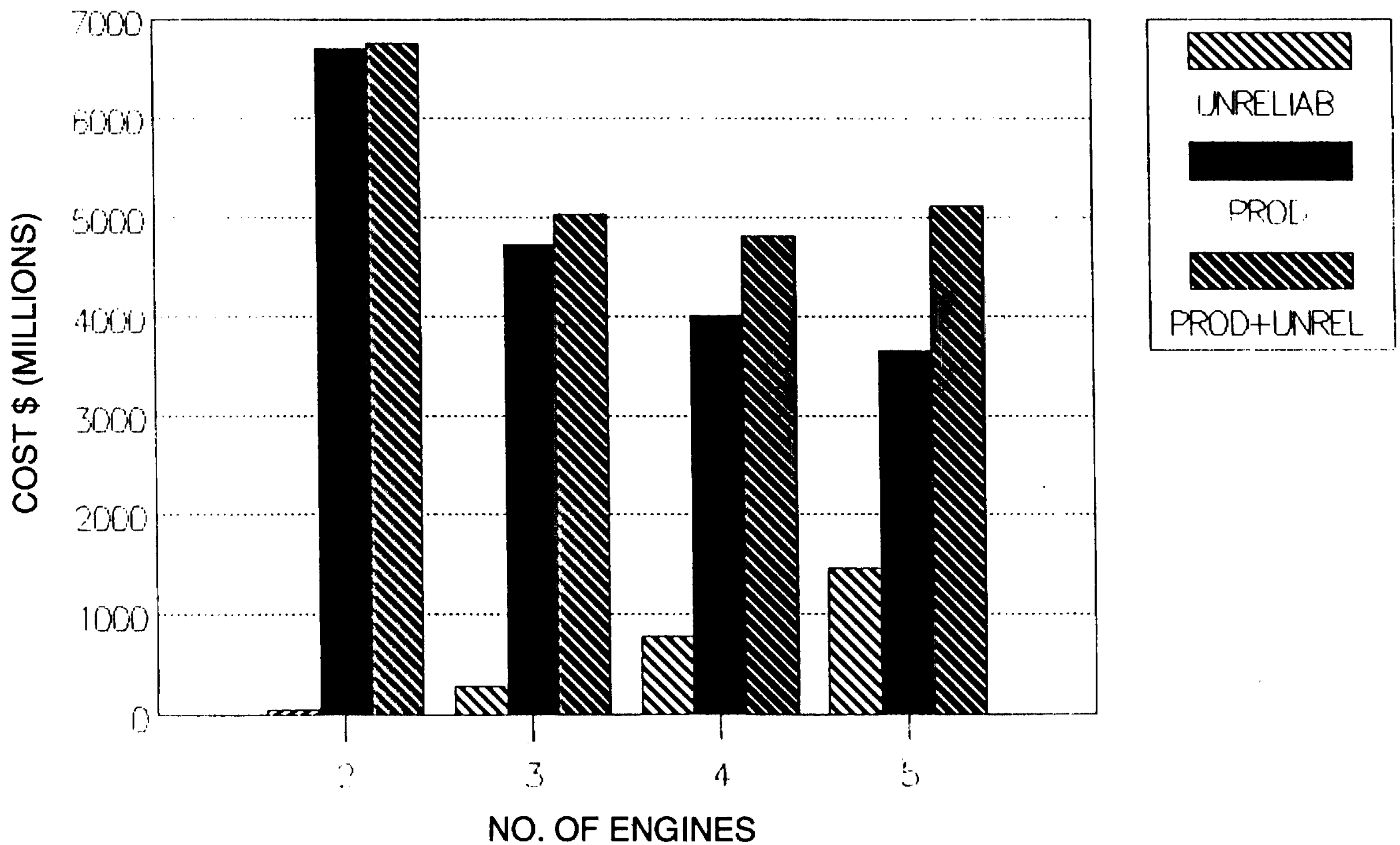
HRB ENGINE RELIABILITY TRADES
SINGLE ENGINE OUT 1/MO LG EXPENDABLE



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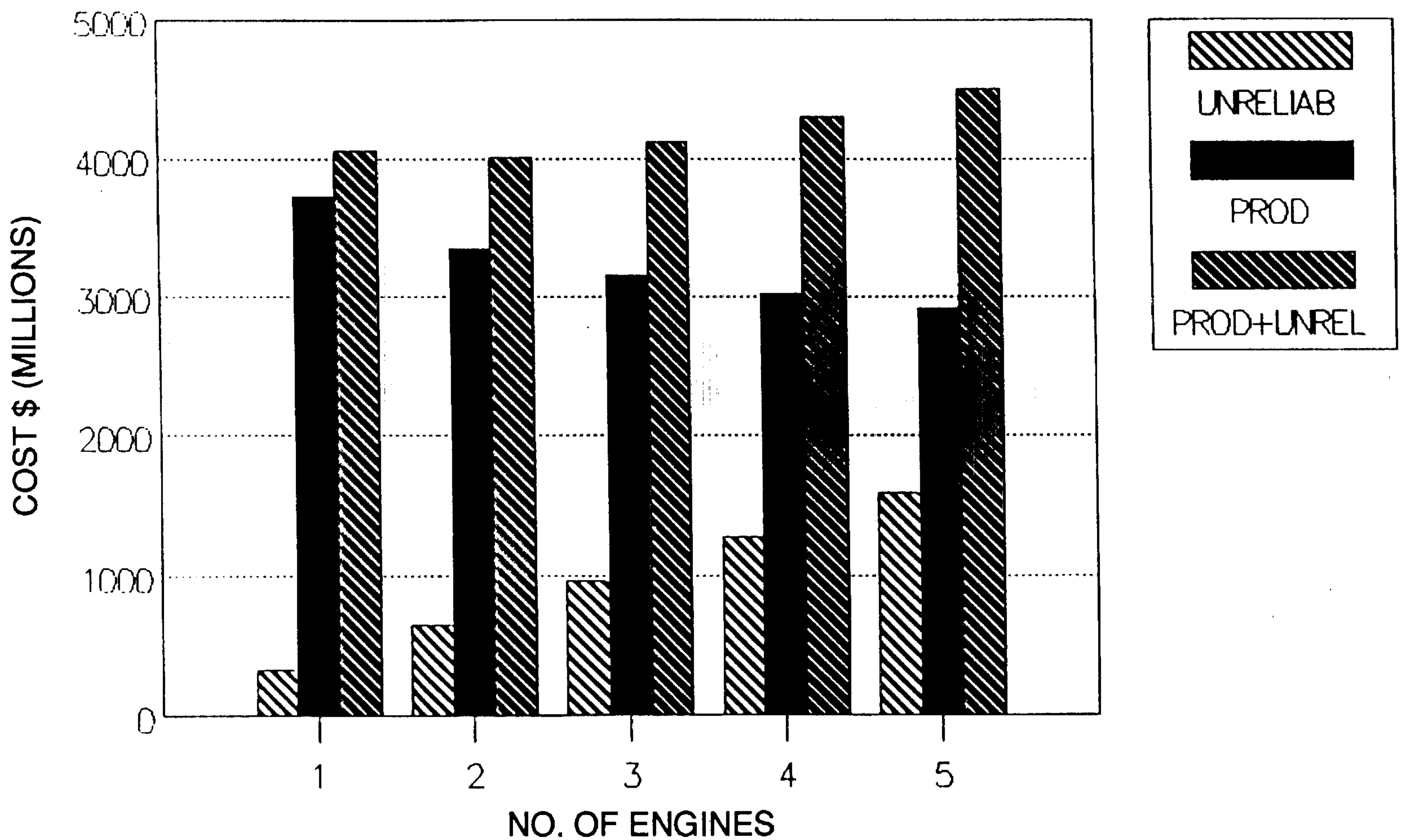
HRB ENGINE RELIABILITY TRADES
NO ENGINE OUT 1/MO LG EXPENDABLE



The Highest Number of TPAs Scored Is Four, as This Scoring Chart Shows. The Maximum Payload Is Carried by Propulsion Having Weight and Cost Turbomachinery

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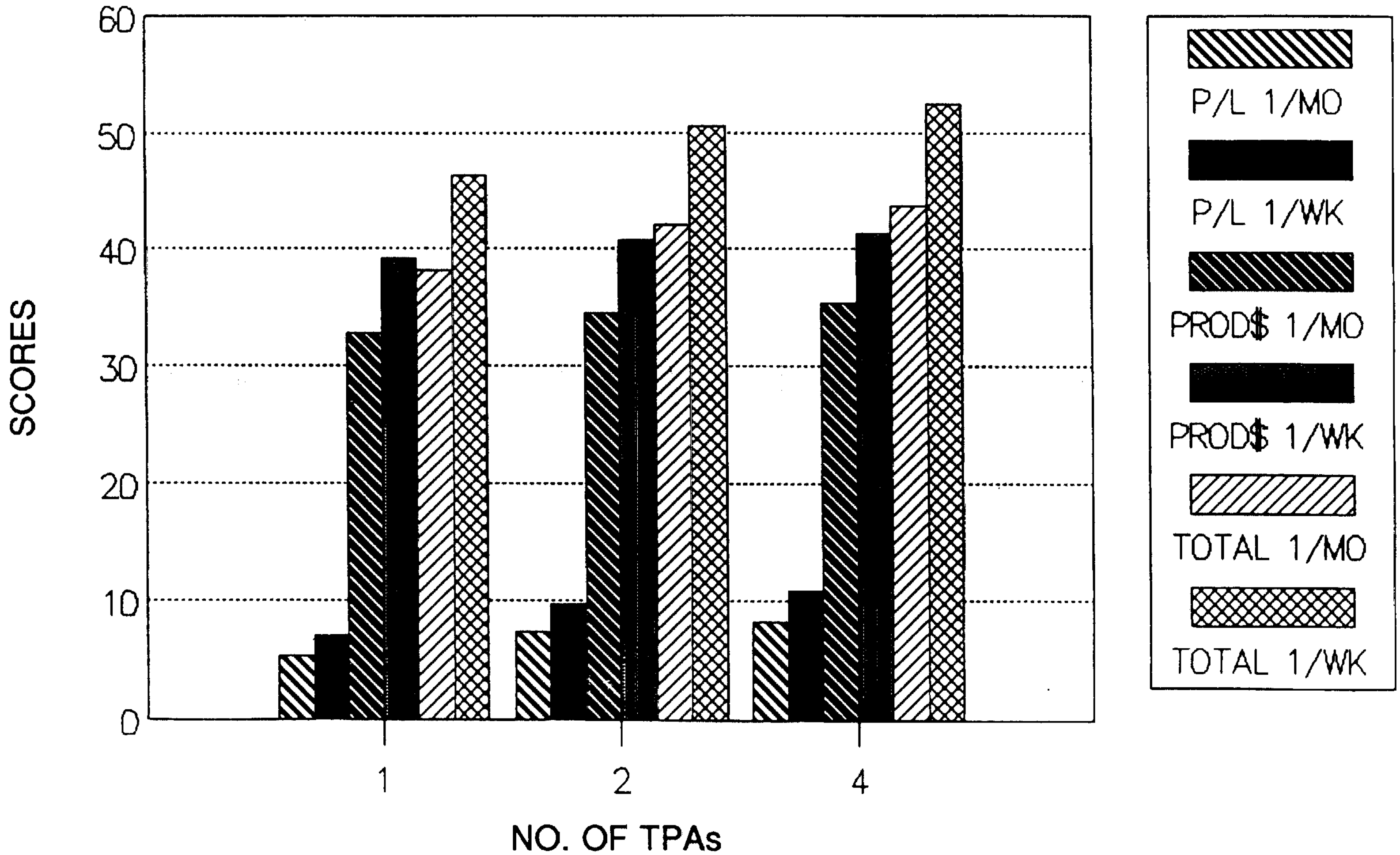
TPA SELECTION SUBSYSTEM OPTIMIZATION



This Chart Shows How TPA Size and Weight Varies With Number/HRB. The Bars Depict the Total Turbomachinery Weight/Large HRB

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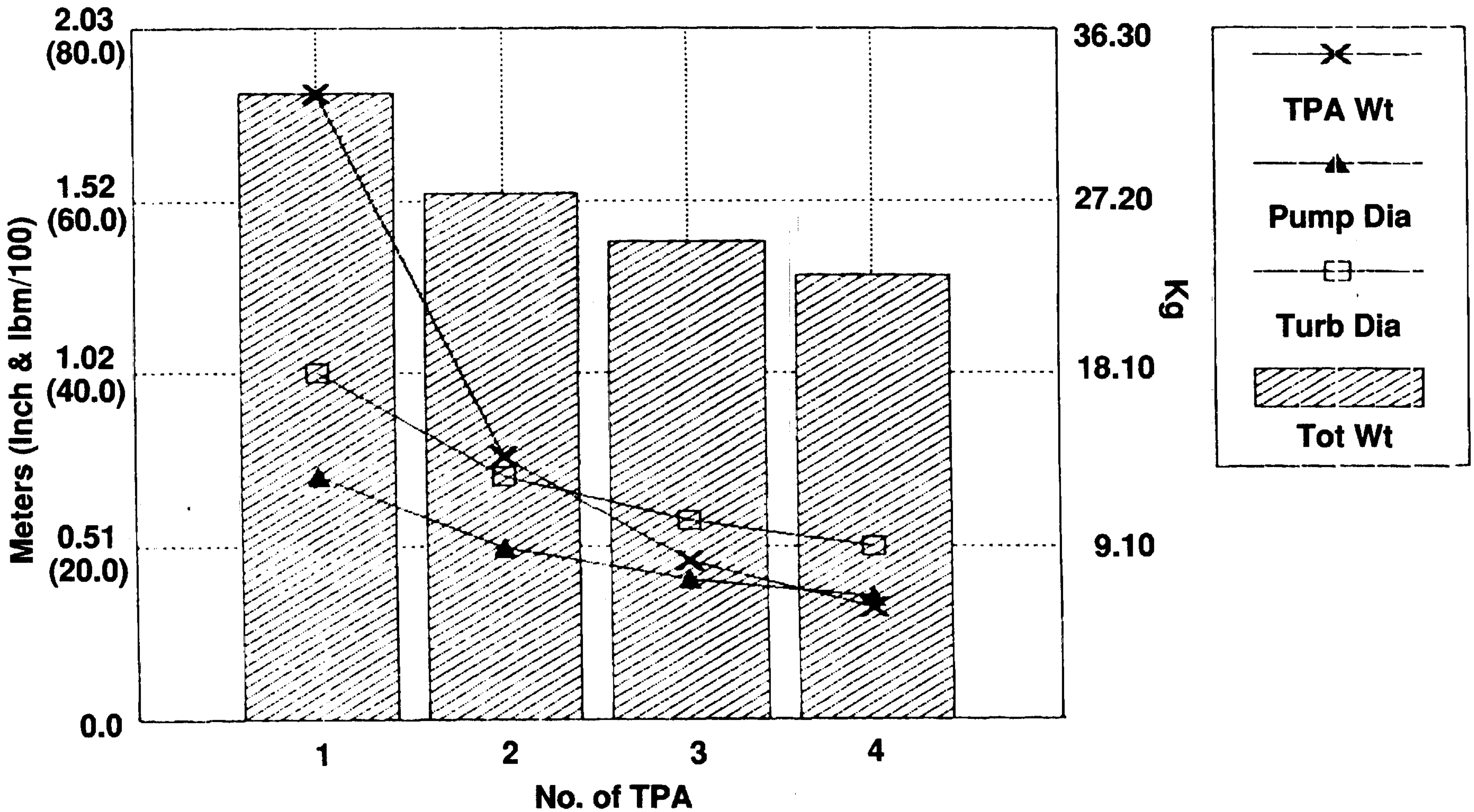
HRB TPA DATA



The Choice of Four Engines (TCA and TPAs) in the Engine Bay of a Large HRB, Requires a Review of How Many Solid Cases Should Be Used, Four or One?

The Chart Below Summarizes the Options in Matrix Form, Including the Four Small HRB Option (at the Right) for Comparison. It Shows the Many Costly Factors That Must Be Added to Large HRBs to Allow Them to Make Mission With a Failed Engine (or TCA) vs the One Item (a Flow Control Plate at the Injector Inlet) Needed to Make Safe Abort. Four Solid Case Large HRB Concept Automatically Makes Safe Abort, Therefore, Why Select the Single Case?

The Reason Is FOM Score. The Single Case Out Scores the Four Case Concept, Because the Latter Is Both Heavier and More Complex Than the Single Case—Leading to Low Production and Payload Scores for the Four Case Concept. Bundling Four Small Units Together Requires More Hardware, and They Are Not as Stiff in Bending Without Shear Load Transmission Between Cases <u>and</u> Heavier Case Walls

The Gas Flow Control Device Is Also Needed to Control HRB Mixture Ratio Independently From Thrust Level, and to Balance Flow Between TCAs. It Appears to Be a New Technology Device Amenable to Development Rather Than Breakthrough

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WE SELECTED A SINGLE MOTOR CASE CONCEPT FOR THE LARGE HRB SCENARIO

Factors	Options		
	Large Diameter Motor Case	Small Diameter Motor Case	
1. Motor Case Size	Solid Burn Rate 2/3 ASRM	Solid Burn Rate ~ 1/3 ASRM	
2. Design Format (2 vs 8 HRBs)	2 HRBs/Launch With 1 Large Motor Case/HRB and 4 TCAs/HRB	2 HRBs/Launch With 4 Small Motor Cases/HRB and 4 TCAs/HRB	8 HRBs/Launch With 1 TCA/HRB
3. TCA-Out Philosophies: A. Make Mission With 1 TCA Out	Yes, If: 1. 4/3 Thrust Engines 2. 2:1 Throttling 3. Gas Shutoff-Without Cat. 1 Safety Problem or	No - Unburnable Solid Propellant or	No - Unburnable Solid Propellant or
B. Safe Abort With 1 TCA Out	Yes, If: 1. Gas Flow Control - No Gas Shutoff	Yes	Yes
4. FOM Score	High Selected Concept Is 3.B. for Large HRB 1. Mission Losses Cost Less Than Larger Engines 2. No Cat. 1 Safety Problem Without Gas Shutoff 3. Gas Flow Control Also Gives: • MRD Control • Independent MR Control	Lower • P/L Loss Due to Drag and Weight • Production Cost up Due to Weight and Complexity	Lower • P/C Loss Due to Drag and Weight • Production Cost up Due to Weight and Complexity

New Technology Required: Warm Gas Flow Control

This Chart Shows the Scores Favoring a Single Solid Case for a Four Engine, Large HRB

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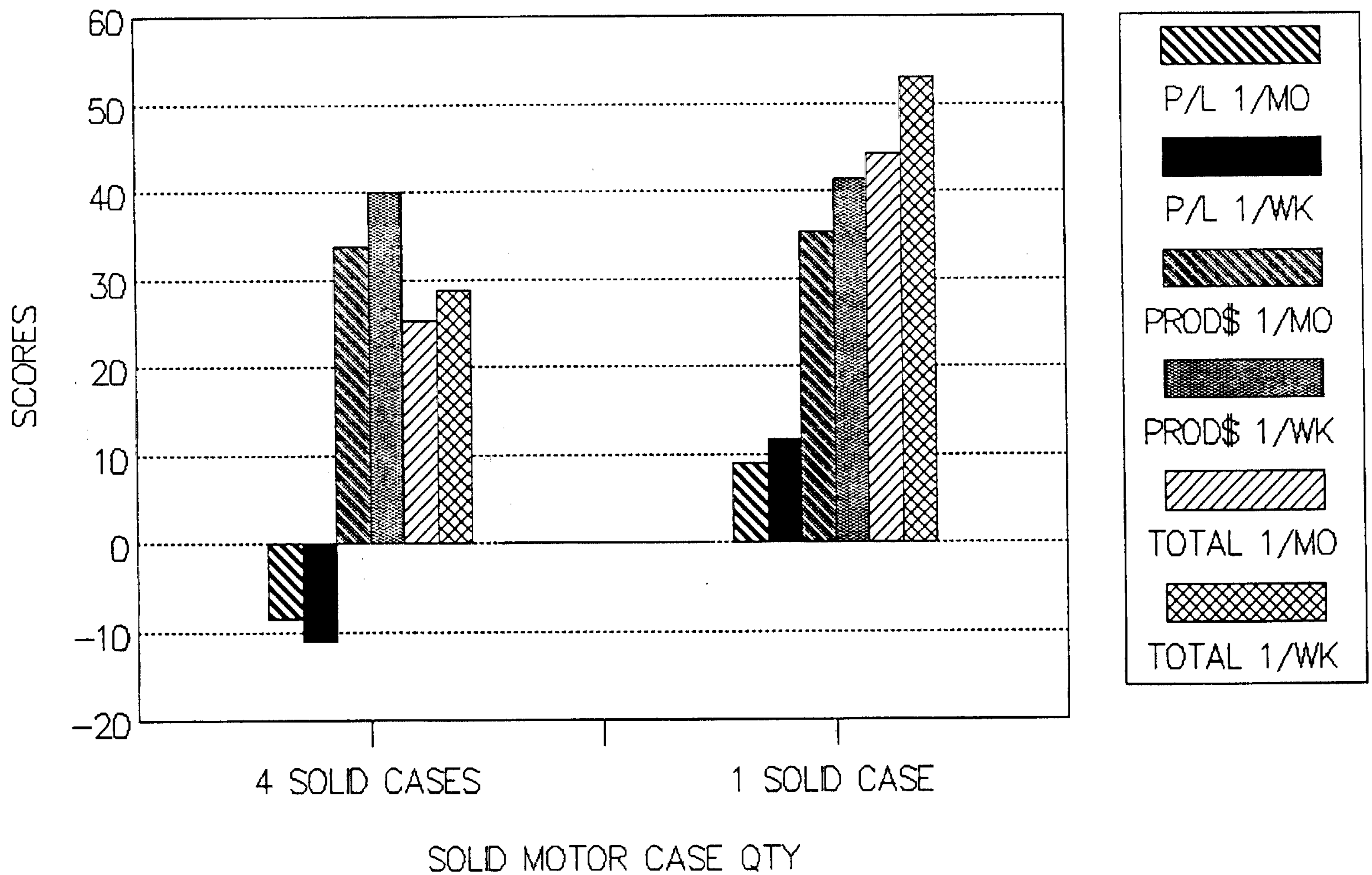
SOLID CASE NUMBER SELECTION DATA LARGE HRB



We Examined HRB Size and Number (Two vs Eight Per Launch) at Both Low and High Flight Rates. Large HRBs Scored Higher for Reasons Similar to Those That Caused Us to Select One Solid Case vs Four for Large HRBs. Greater Weight and Complexity Gives Lower Scores for Multiple HRBs

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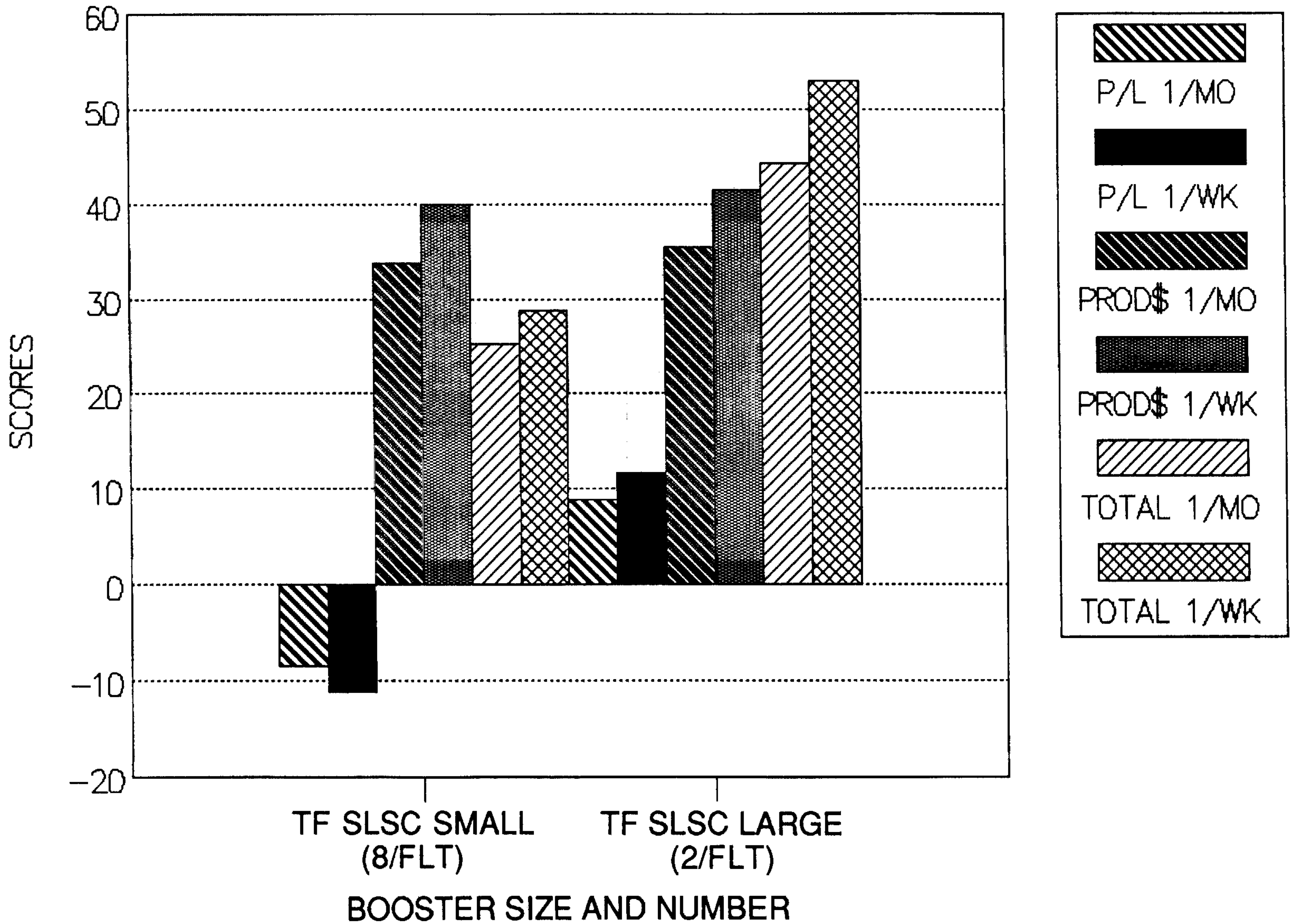
HRB NUMBER AND SIZE SELECTION DATA



BOOSTER REUSE SCENARIOS

Concept	Screening Result
Flyback Booster	No - Not Compatible With STS
Recover Booster	Yes - Study FOM Score
Recover Engine Module	Yes - Study FOM Score

As the Chart Shows, We Examined Recovery of Both Pressure Fed and Turbopump Fed Large HRBs at Low and High Flight Rates. We Found That for the Lower Rate of Once Per Month Launches There Was No Discernible Score Difference Between Reused or Expended HRBs for Pump or Pressure Fed Concepts. At the Higher Flight Rate (and More Total Flights), However, Both Concepts Scored Higher as Expendable HRBs. This Happens, Because With More Flights the Greater Payload-Carrying Benefit of the Expendable Systems (No Recovery Systems) Become Dominant

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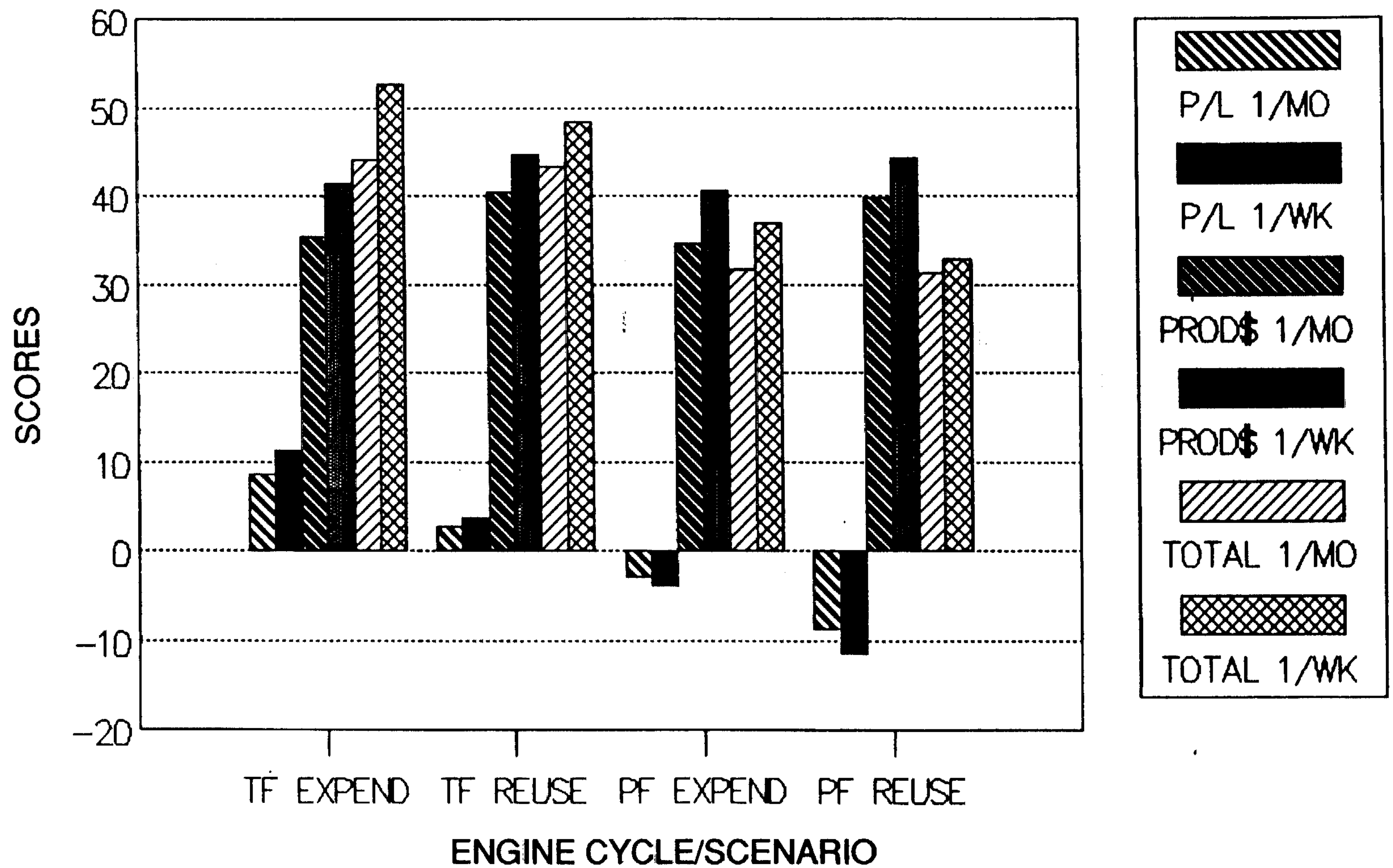
HRB REUSE SELECTION DATA LARGE HRB RECOVERY SCENARIO



We Examined Recovery of Both Pressure Fed and Turbopump Fed Engines and Avionics in a Module at Both Low and High Flight Rates. We Expected Better Reuse Scores With This Scenario, Because Recovery Is Limited to a Relatively Small, Lightweight Module of High Value Per kg (lb) as Compared to an Entire Expended HRB

Our Design Is Such That All Engines Are Hard-Mounted to the Aft Motor Case Head, Which Attaches Below the Aft Skirt Joint. Separating This Joint, the LO_2 Feed Line, the Tank Pressurization Line, and the Ablative Engine Nozzles Below Their Rubber Flex Joints (After HRB Separation From the ET) Provides a Pancake Module of About 2.75 m (9 ft) High and 4.875 m (16 ft) in Diameter. It Has a Parachute and Inflatable Flotation Pack Located at the Center of Its Flat Nozzle and Heat Shield End. The Module Will Land in the Water With Engine Nozzle Upward and Float Inside Its Inflated Toroidal Collar

Our Study Uses ALS Contractor Data for Module-Recovered Engine Refurbishment Costs. It Shows That the Engine Module Recovery Reuse Scenario Has Higher Scores Than Expendable Systems for Turbopump Fed HRBs by 5 to 10%, Whereas There Is Little Reuse Advantage for Lower Engine Cost Pressure Fed HRBs. Refurbished Pressure Fed HRBs Need 65% New Hardware; Turbopump Fed HRBs Need Only 42% New Hardware (by Cost). Refurbishment Costs Are 27% of New Hardware

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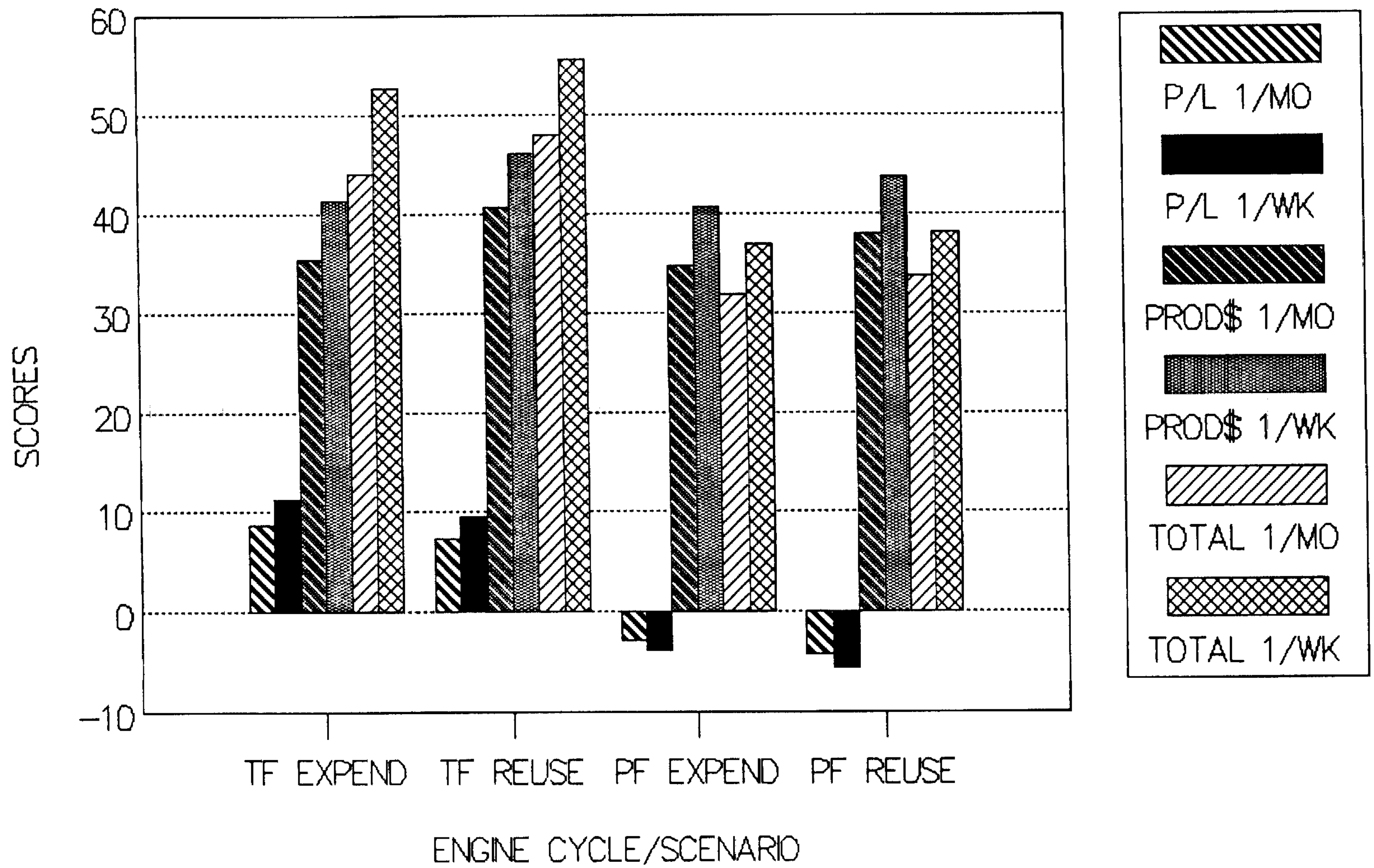
HRB REUSE SELECTION DATA ENGINE MODULE RECOVERY SCENARIO



Hybrid/Pt 11/ 8

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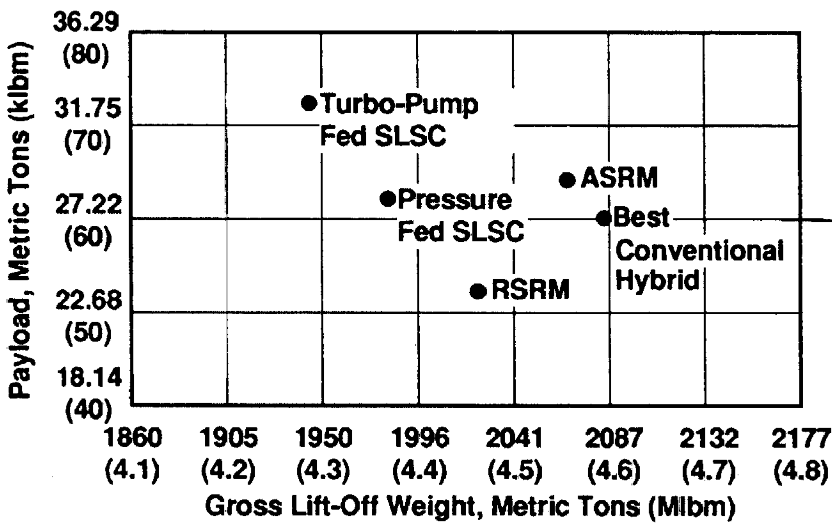
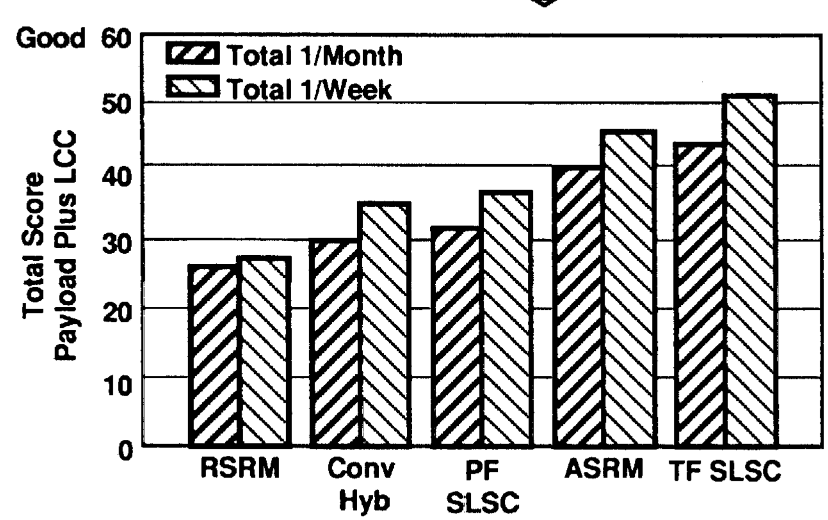
WE HAVE APPLIED OUR SCORING TO EXISTING AND FUTURE STS BOOSTERS





The Turbo-Pump FED SLSC Hybrid is a Logical Growth Path to Provide:

- Flight Safety and Reliability
- Low LCC
- Performance

With:

<u>Abort</u> and <u>Throttling</u>

182-1

M13 HD-039

In Task 4, Conceptual Design Definition, We:

- Performed Payload and Weight Optimization Studies of,
- Performed Sensitivity Studies on,
- Performed Supporting Analyses of, and
- Prepared Design Layouts and Specifications for Our Selected HRB to Create a Concept Definition Package of the Best HRB

We Will Begin by Showing Our Design and Specification First, Followed by Component Designs and Specifications, and Supporting Analytical Results. Lastly, We Will Show the Results and Design Decisions From Our Design Sensitivity and Optimization Studies

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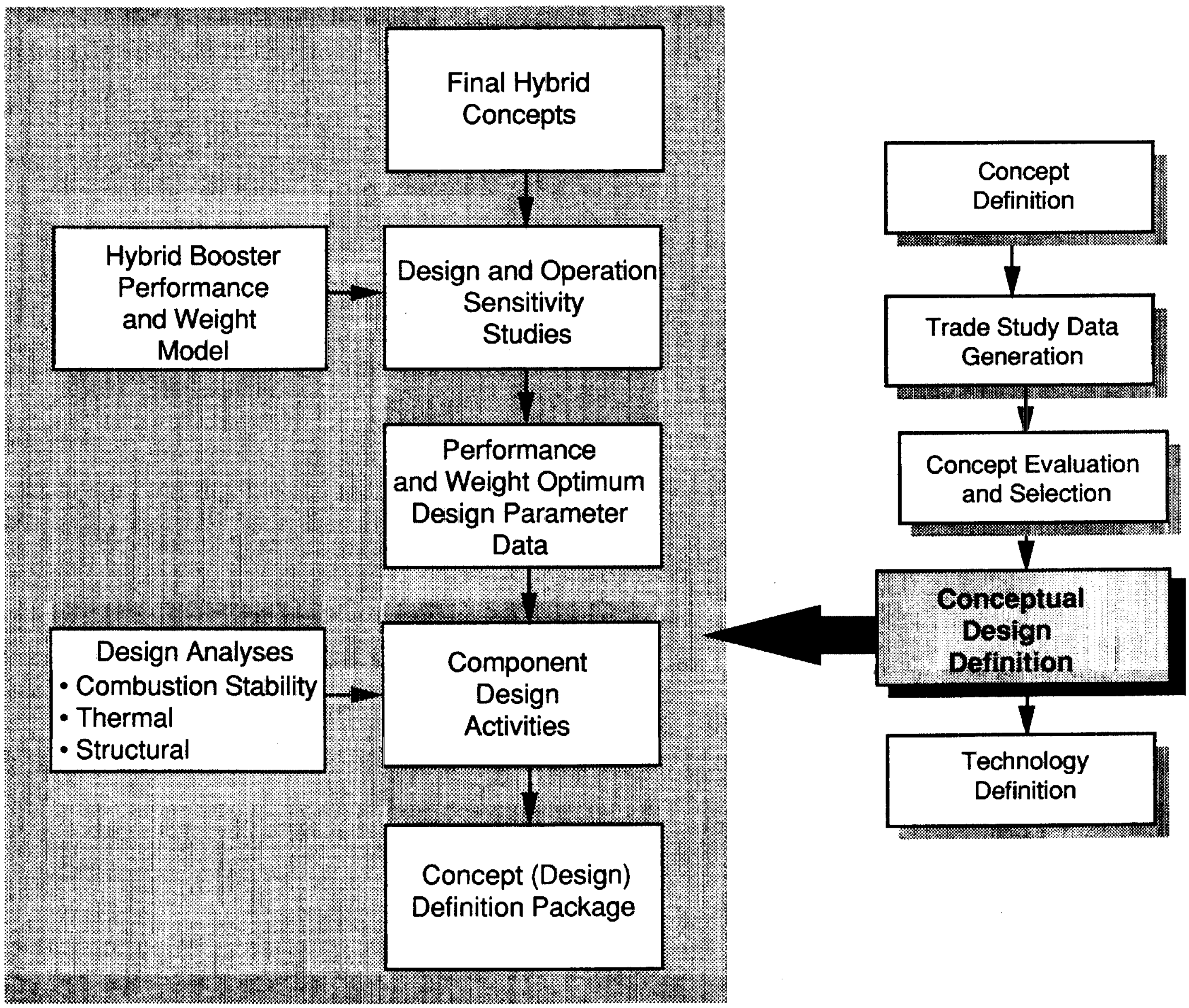
OUR CONCEPTUAL DESIGN DEFINITION PROCESS MAXIMIZES PERFORMANCE AND MINIMIZES DEVELOPMENT RISK



HRB DESIGN

Our HRB Design Adheres to Our Design Philosophy. It's Safe and Reliable, Because We Pay Attention to Critical Failure Potentials at the Earliest Point in the Design Process, and We Demonstrate All Technologies in Phase II Before Proceeding. We Use Systems Models From the Beginning, and Perform Variance Analyses to Create Robust Designs That Are Controllable and Do Not Degrade Easily in Response to System Changes. We Build in Reliability by Allocating Requirements and Assuring That They Can Be Met

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HRB FOR STS DESIGN PHILOSOPHY— SAFETY AND RELIABILITY

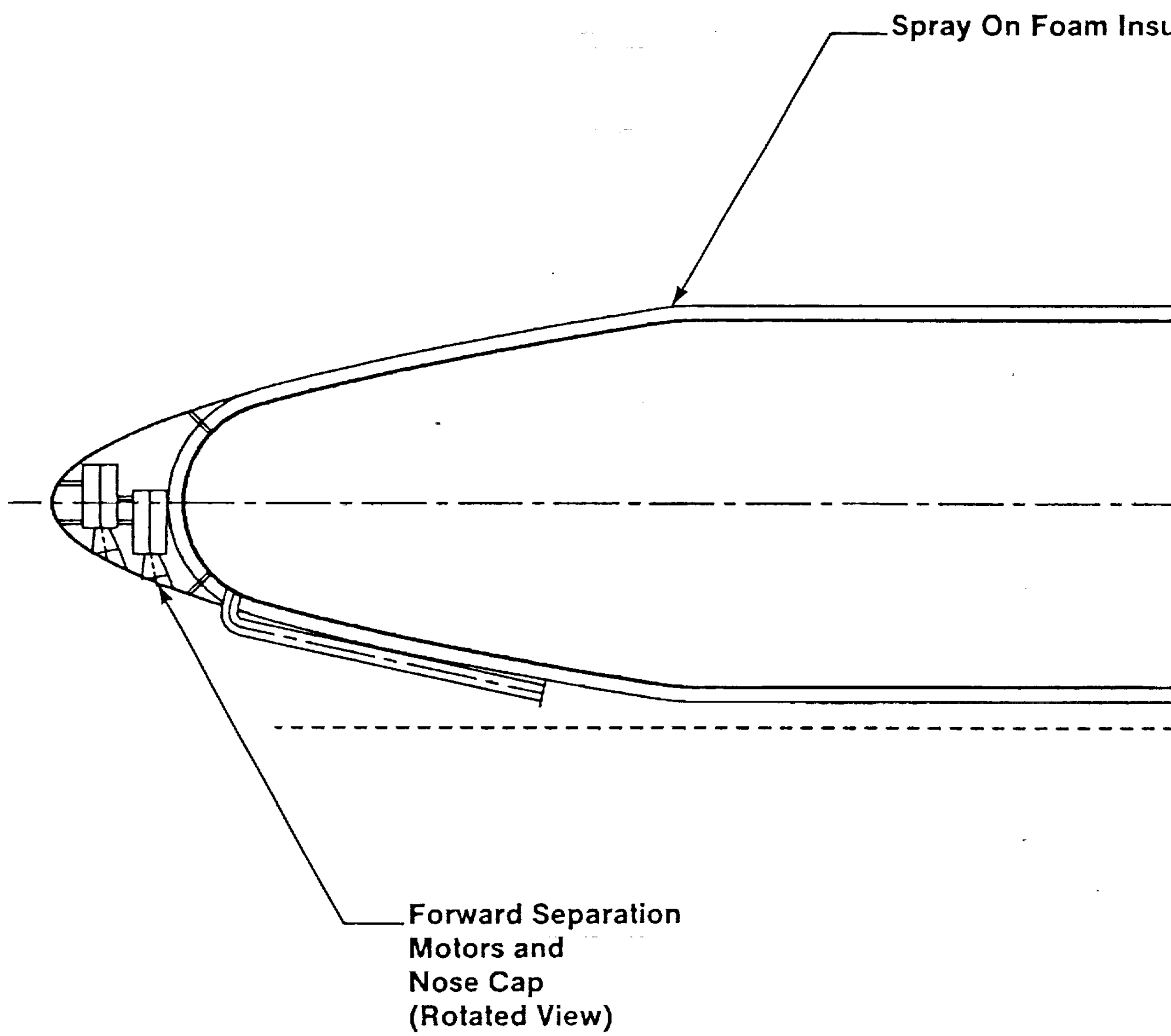
- Create Safe Design
  - FMEA/CIL Studies
  - Use Demonstrated Technologies
  - Use Historical Failure Data
- Create Robust Design
  - Variance Analyses
  - System Models
- Create Reliable Hardware
  - Allocate Reliability Requirements
  - Loads and Properties Analyses
  - Testing and QA
- TQM Philosophy Throughout
  - Early Attention to Safety and Reliability, Cost, and Performance - Simultaneous Engineering
  - Continual Improvement

Our HRB Design Has Many Safety, Cost, and Payload Features, and They Are Shown on Our Layout Drawing

1.

FOLDOUT FRAME

SLSC HRB FEATURES



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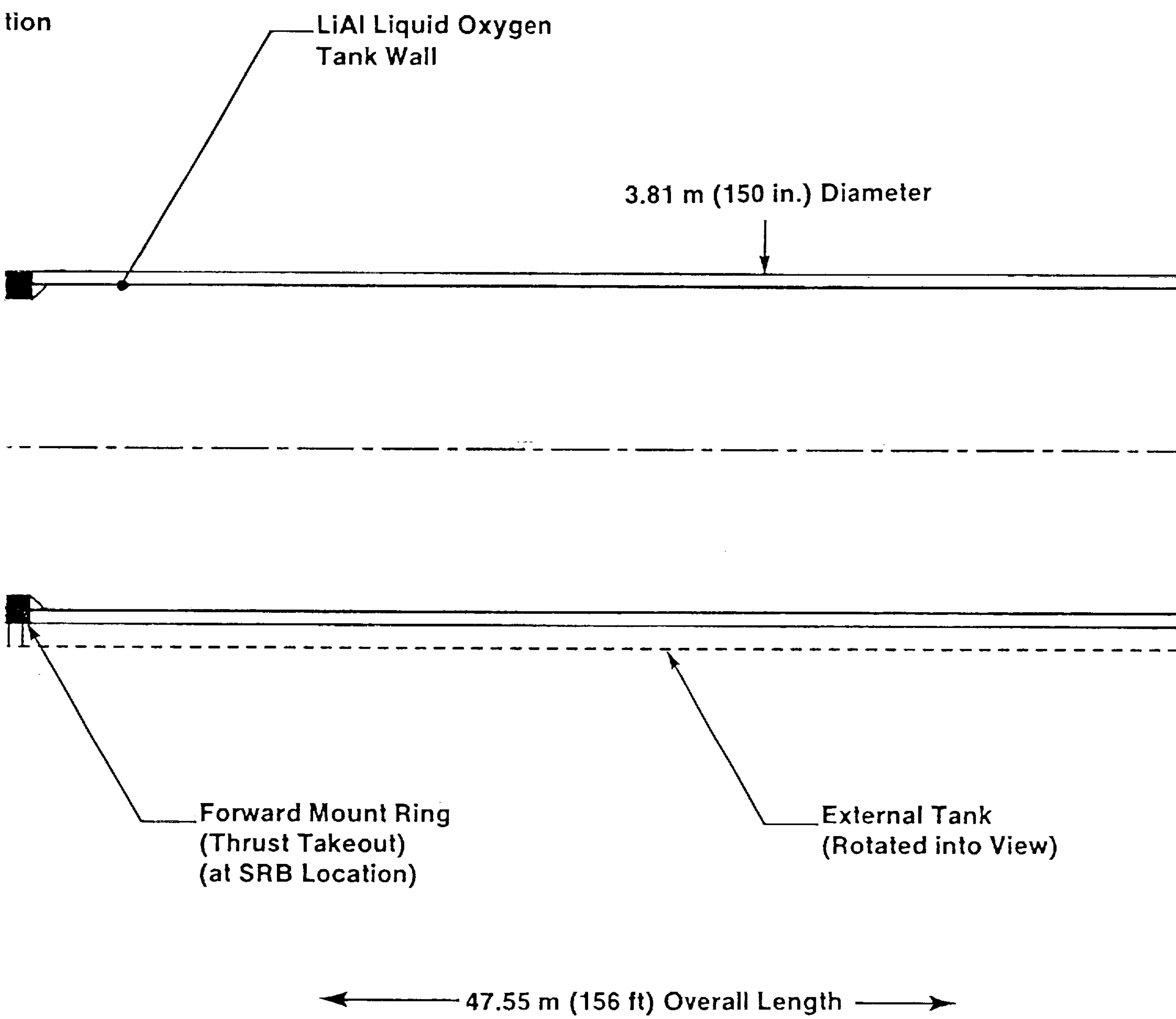
Our HRBs Attach to the ET With Rings and Struts at the Same Locations as SRMs Do. Our Four Engine Nozzles Will Fit Between MLP Support Posts When the Posts Are Moved Into a Square Pattern. A Single Feed Line Brings LO_2 From the Low Point of the Tank During Flight to the Engine Bay and Routes to Four Hard-Mounted TPAs Through Simple and Symmetrically Repetitive Hard Lines. The Two HRBs Are Identical; Only the Mount Struts Are Relocated

1

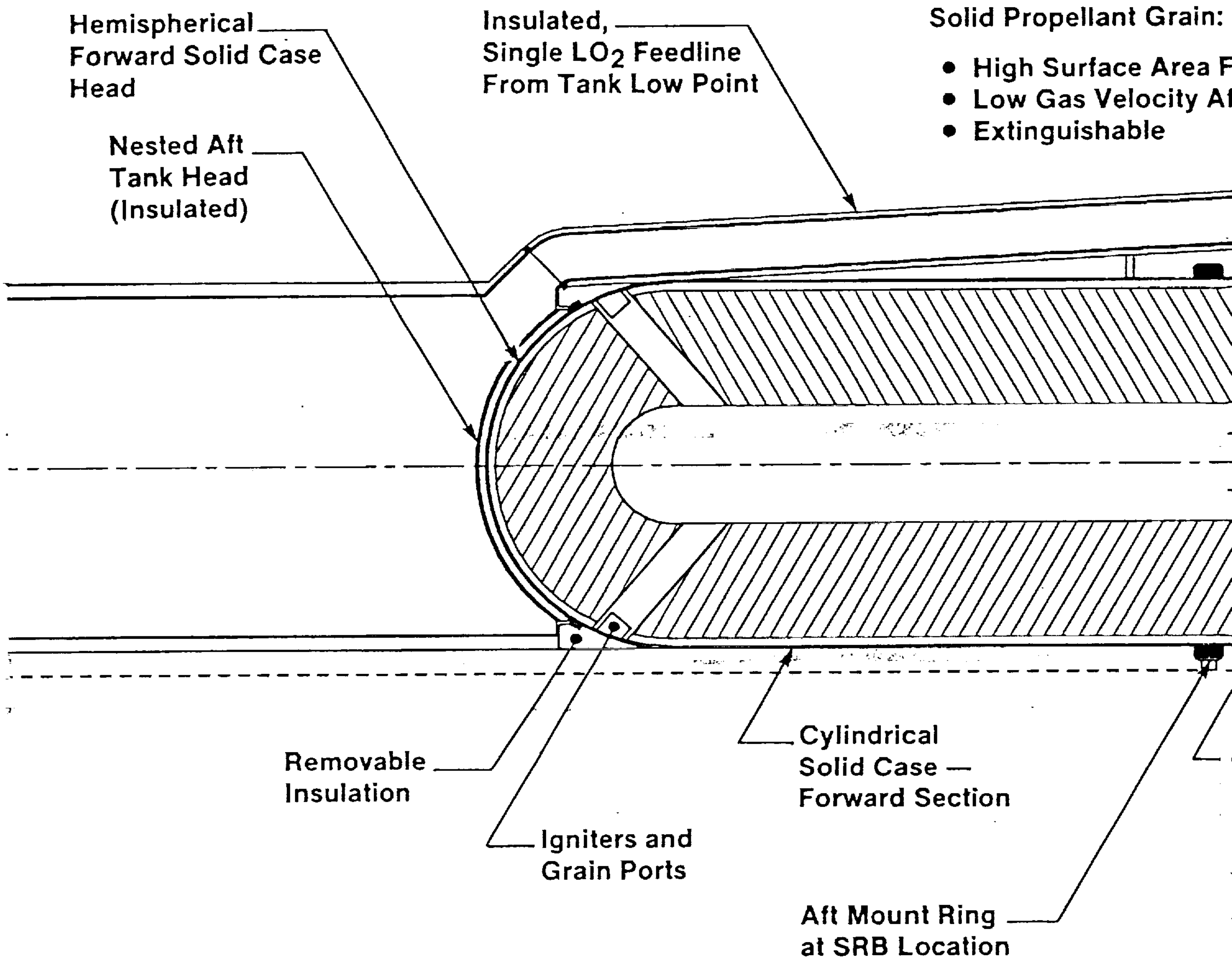
2

FOLDOUT FRAME



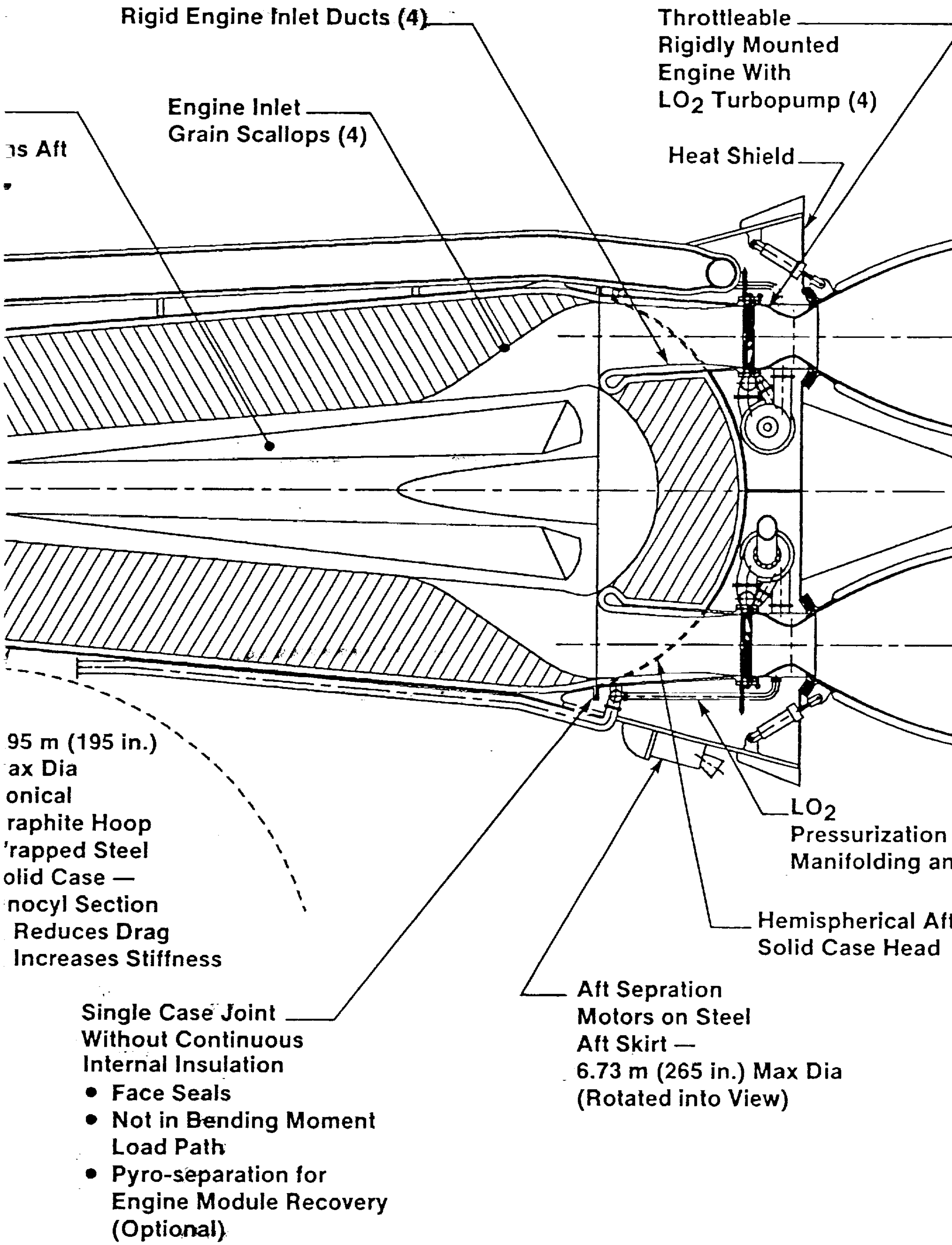
1

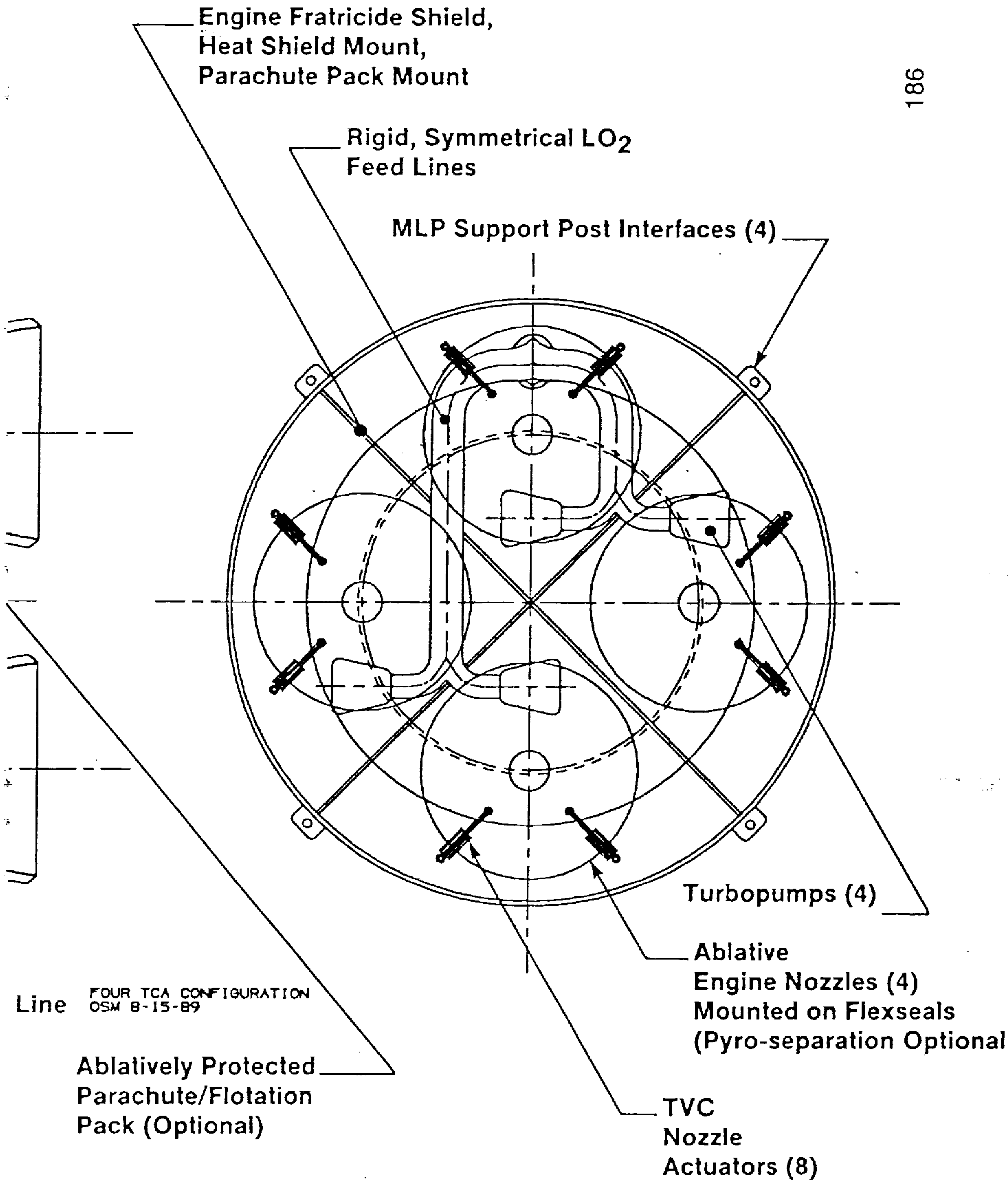
FOLDOUT FRAME 3 -



4.

FOLDOUT FRAME




Engine Fratricide Shield,
Heat Shield Mount,
Parachute Pack Mount
Rigid, Symmetrical LO_2
Feed Lines
MLP Support Post Interfaces (4)
Turbopumps (4)
Ablative
Engine Nozzles (4)
Mounted on Flexseals
(Pyro-separation Optional)
Line
FOUR TCA CONFIGURATION
OSM 8-15-89
Ablatively Protected
Parachute/Flotation
Pack (Optional)
TVC
Nozzle
Actuators (8)

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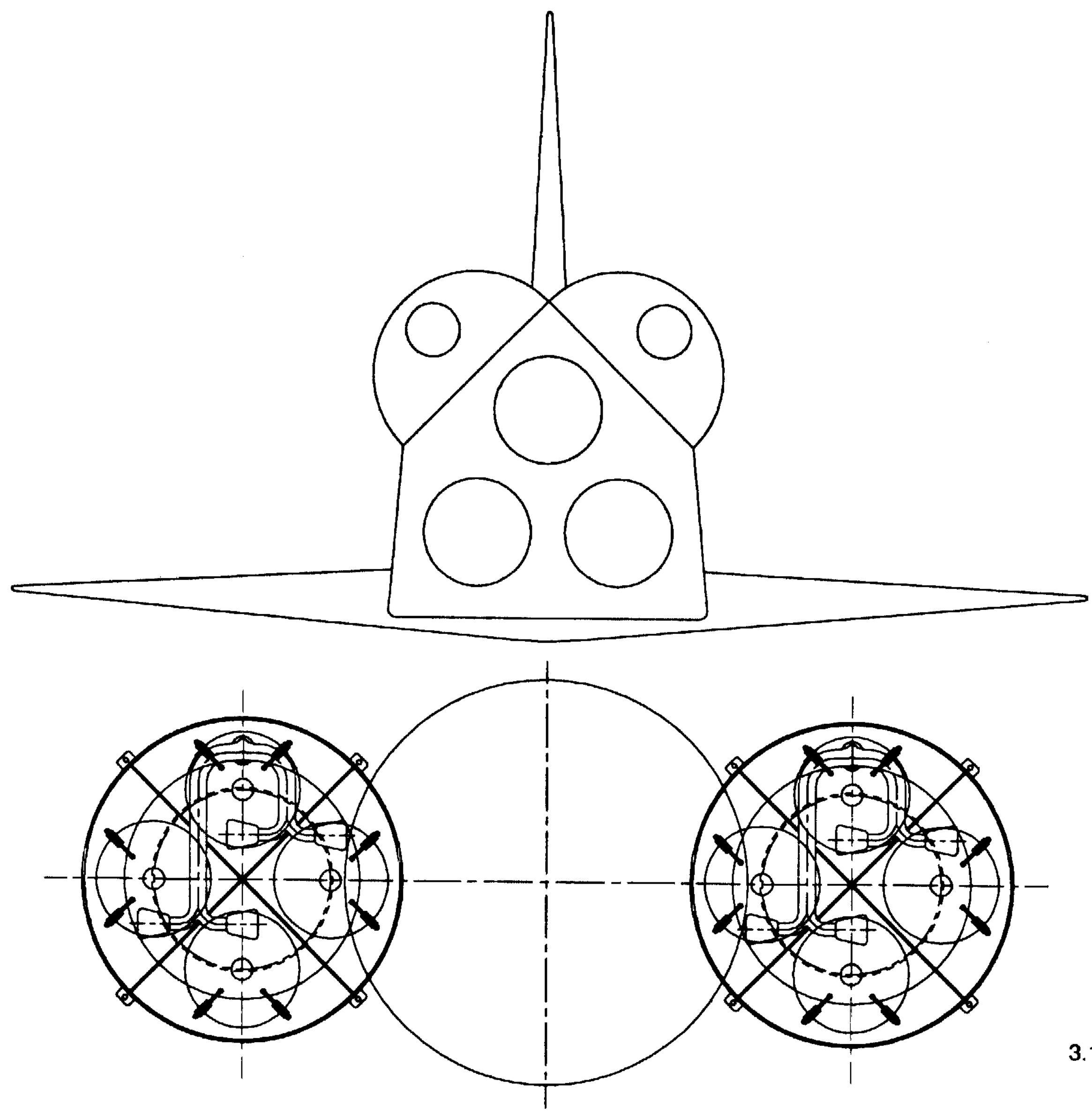
SPACE TRANSPORTATION SYSTEM VIEW

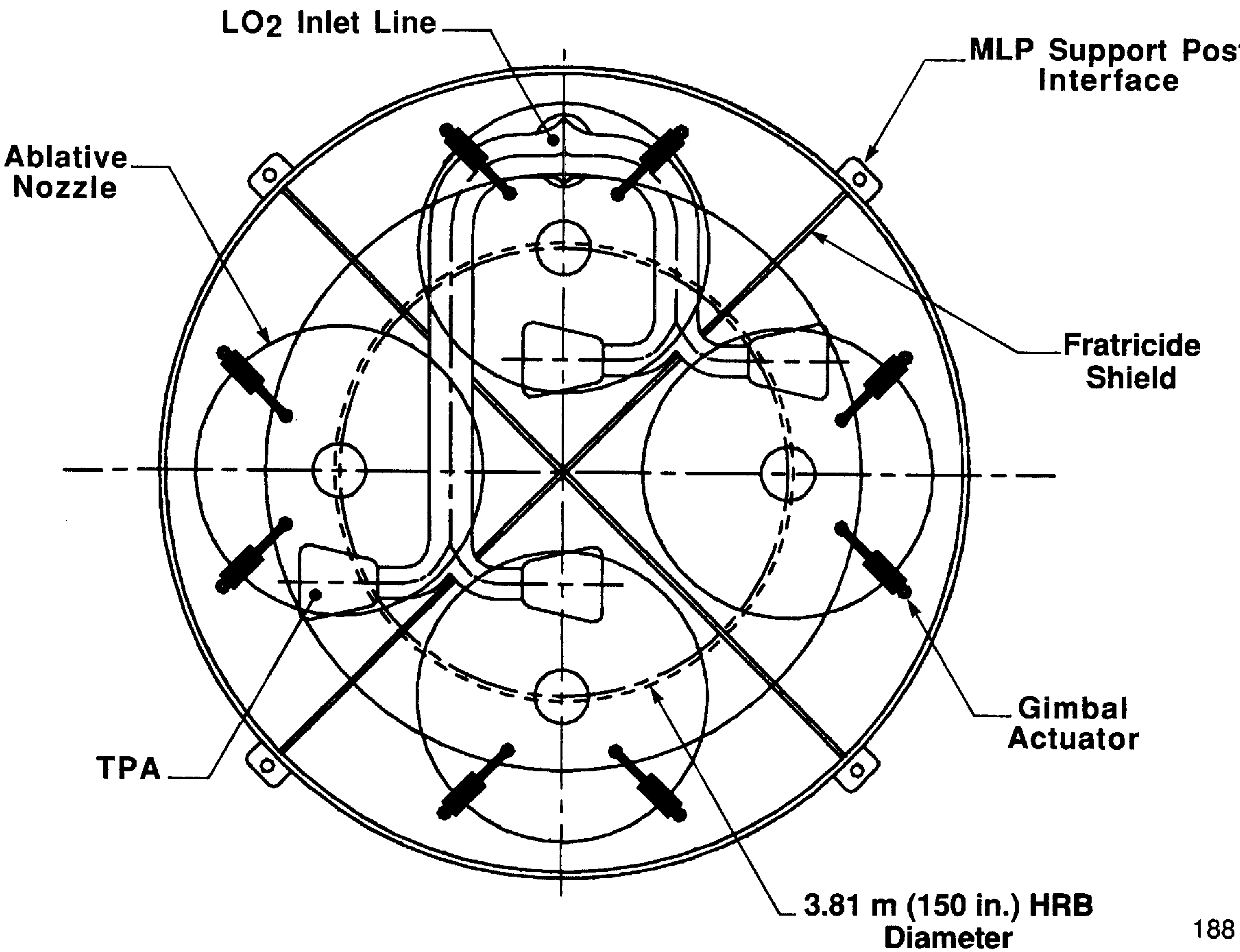


3.16.0.17

The Engine Bay Is Divided Into Four Compartments by Intersecting Steel Plates From the Aft Solid Case Head Past the Thrust Chamber and Turbopumps. The Gusset-Type Plates Function as Fratricide Shields to Improve Booster Reliability, Heat Shield Mount, and Optional Recovery Pack Mounts. LO_2 Lines Pass Through the Plates, Mount to Them, and Deliver Oxidizer to Four Side-Mounted Turbopump Inlets. Flow Straightening Vanes May Be Used to Improve LO_2 Inlet Flow Symmetry

Each Engine Is Identical; Two Are Rotated 180 Degrees From the Other Pair. Each Flexseal-Mounted Supersonic Rocket Nozzle Is Controlled by Two Actuators, Mounted to the Aft Skirt. They Act in Unison to Provide Pitch and Yaw Motion, and Nozzle Motion Will Be Less Than the Required Gimbal Angle by About 20 to 40%. This Benefit Derives From the Fact That Supersonic Splitline Nozzles Achieve TVC Amplification Factors That Are Caused by Shock Wave Induced Pressures Creating Side Loads on the Nozzle Walls. This Effect Is Quantified on the Next Page

AFT VIEW


LO2 Inlet Line
MLP Support Post Interface
Ablative Nozzle
Fratricide Shield
Gimbal Actuator
TPA
3.81 m (150 in.) HRB Diameter

Atlantic Research Corporation and Aerojet Have Investigated Supersonic Splitline TVC Experimentally. Amplification Factor Is Defined as Degrees of TVC Achieved Divided by Degrees of Nozzle Movement. Approximately 6 Degrees of Movement Will Achieve 8 Degrees of TVC

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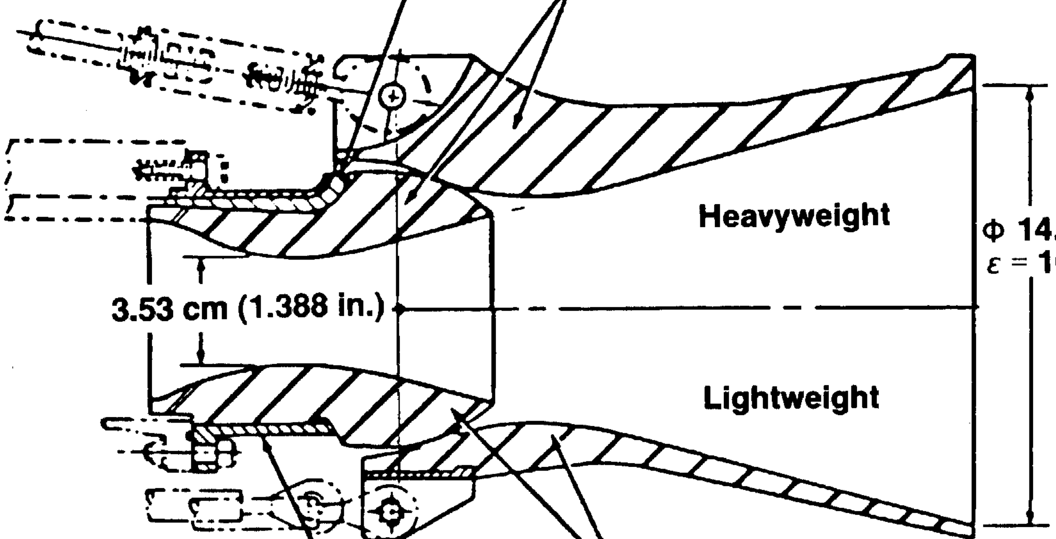
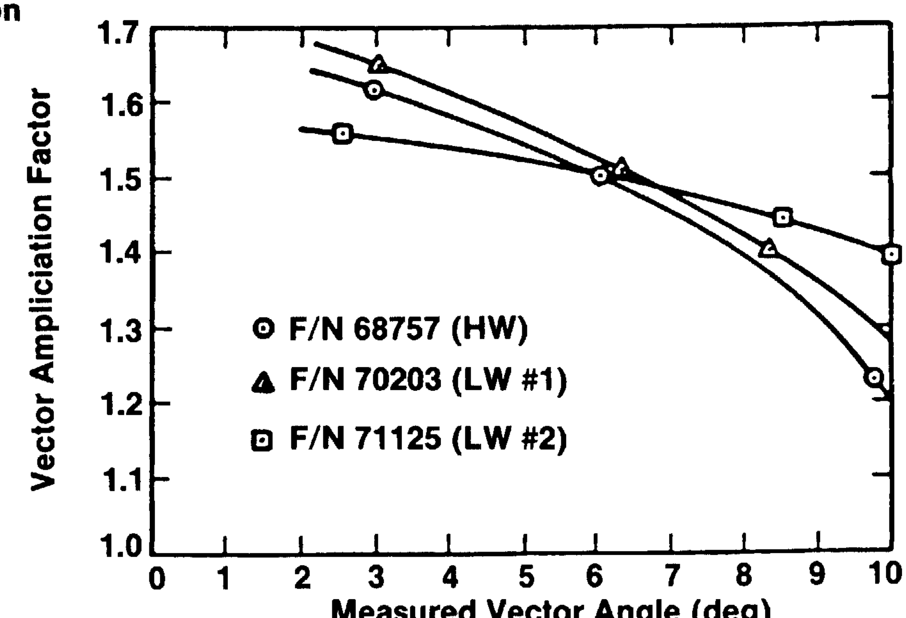
TVC AMPLIFICATION FACTORS ARE LARGE FOR SUPERSONIC SPLITLINE NOZZLES



Supersonic Splitline TVC Nozzle



Achieved Vector Amplification For 3 Firings

Our HRB Operational Sequence Is as Follows:

- Chill Down and Bleed in the LO_2 Pump and Injector With Bleed Valve
- Open Facility GN_2 Valve to Spin Turbopump With Turbine Bypass Valve Closed. GN_2 Exhausts to Rocket Nozzle at $\varepsilon = 3.0$
- Ignite Solid Propellant Grain
- Combustion in Thrust Chamber Begins When LO_2 and Solid Grain Fuel Rich Warm Gases Meet. LO_2 Bleed Flow in Regenerative Cooling Jacket Receives Heat
- Turbine Receives Heated O_2 and Flashes to GO_2 Drive Fluid in Nozzles. GO_2 Turbine Exhaust Follows N_2 Into Burnoff Manifold at $e = 3$ in Rocket Nozzle and Forward to LO_2 Tank Ullage at 1724 kPa (250 psia). Fuel Rich Boundary Layer Burns Off in Nozzle With GO_2 Turbine Exhaust
- LO_2 System Bootstraps as Solid Grain Fully Pressurizes. Remove Facility GN_2 Line
- Thrust is Controlled With Turbine Bypass Valve That Presents Loss of Regenerative Coolant Flow
- O/F Mixture Ratio Is Controlled With Flow Control Plate Forward of Gas/Liquid Injector
- TPA Provides Alternator Power for Valve and TVC Actuators. Ablative Nozzle Is Attached With a Flexseal
- Near End of Operation the LO_2 Ullage Pressure Control Valve Is Closed to Let Ullage Pressure Drop to Reduce Tank Weight at Burnout and Tank Stiffness
- Shutoff LO_2 When Staging and Open Control Plate to Extinguish Solid Propellant

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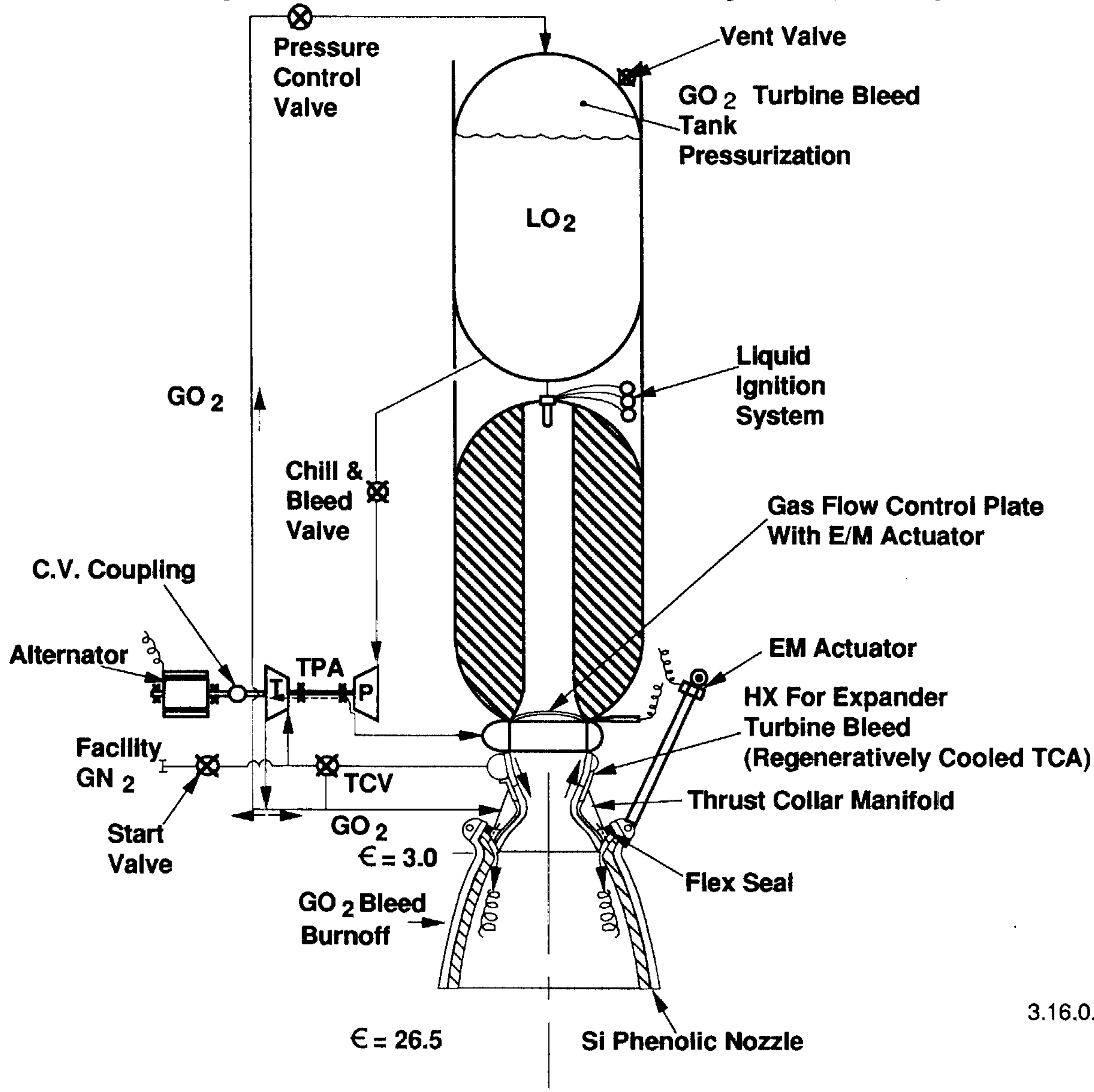
OUR HRB FOR STS TURBINE DRIVE CYCLE IS SELECTED

Expander Bleed Burnoff Cycle (EBB)



3.16.0.9

We Have Prepared a Preliminary Propulsion System Instrumentation List, Including Performance, Control, and Health Monitoring Functions

PUMP FED SLSC PRIMARY SYSTEMS INSTRUMENTATION

Location	Measurement	Type
Thrust Chamber Assembly	Chamber Pressure	I,C,H
	Chamber Vibration	I,C,H
Motor Case	Case Pressure - Upper/Lower	I,C
	LOx Inlet Flow	I,C
	LOx Inlet Pressure	I,C
LOx Tank	Ullage Pressure	I,C
	Ullage and Liquid Temperature	I
	Liquid Level - Upper/Lower	I,C
	GOx Inlet Flow	I,C
	Vent Nozzle Pressure	I
LOx Turbo Pumps	LOx Inlet Pressure (2)	I,C
	LOx Inlet Flow	I,C
	LOx Exit Pressure	I,C
	Turbine Inlet Temperature (2)	I,C,H
	Turbine Inlet Pressure (2)	I,C
	Turbine Inlet Flow (2)	I,C
	Turbine/Pump Speed, RPM (2)	I,C,H
	Turbine/Pump Case Vibration (2)	I,C,H
	Turbine Exit Temperature (2)	I,C,H
	Dump Exit Pressure	I

I - Instrumentation for Performance
C - Control Instrumentation
H - Health Monitoring Instrumentation

MARTIN MARIETTA

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AEROJET HRB TF ENGINE CONCEPT DESIGN SPECIFICATION SUMMARY

- Propellants: Fuel Grain No. 8 Sat. HC [PEBC] and LO_2
- Total [4 TCAs] MPL Thrust at Sea Level, 12.24 MN (2.75247 Mlbf)
  MPL Thrust at Vacuum, 14.01 MN (3.14874 Mlbf)
- Combustion Scheme: Solid/Liquid Staged Combustion (SLSC)
- Turbopump Drive Cycle: Expander Bleed Burnoff Cycle (EBBC)
- Gaseous O_2 Autogenous LO_2 Tank Pressurization From Turbine Exhaust
- Dual Ignition System—Oxidizer Rich Liquid or Solid at Forward End of Grain
- Electromechanical-Actuated TVC System With Flexseal Mounted Nozzle
- Turbopump Driven Alternator
- All Hard Feed and Pressurization Lines

AEROJET HRB TF ENGINE CONCEPT DESIGN SPECIFICATION SUMMARY (Cont.)

- Solid Case Aft Head Is Engine Recovery Module Structure
- Design Point ⊗:
  - MPL TCA Pc, 11.72 MPa (1,700 psia)
  - Nozzle Area Ratio, 26.2 - Rao Contour
  - MPL Exit Pressure, 41.37 kPa (6.00 psia)
  - Throat Diameter, 45.7 cm (18 in.)
  - Exit Diameter, 233.7 cm (92 in.)
  - Combustion Mixture Ratio (CMR) 2.60
  - Liquid/Solid Mixture Ratio (LSMR) 1.90
  - MPL I_{sp}VAC and I_{sp}SL, 303 and 265 sec
  - 4 TCAs/TPAs Total Design Weight, 8346 Kg (18,400 lbm)
  - LO_2-Cooled Thrust Chamber
  - Silica Phenolic/Nonmetallic Honeycomb Nozzle, GO_2 Cooled at $\varepsilon = 3$
  - Turbine Inlet Pressure and Temperature, 11.31 MPa and 478°K (1,640 psia and 860°R)
  - LO_2 Pump Outlet Pressure, 14.07 MPa (2,040 psia)
  - Solid Grain MPL Pressure, 12.89 MPa (1870 psia)

Below Is a View of the HRB Engine Bay, Showing the Relationships of the Several Engine, Solid Case, and Interfacing Components

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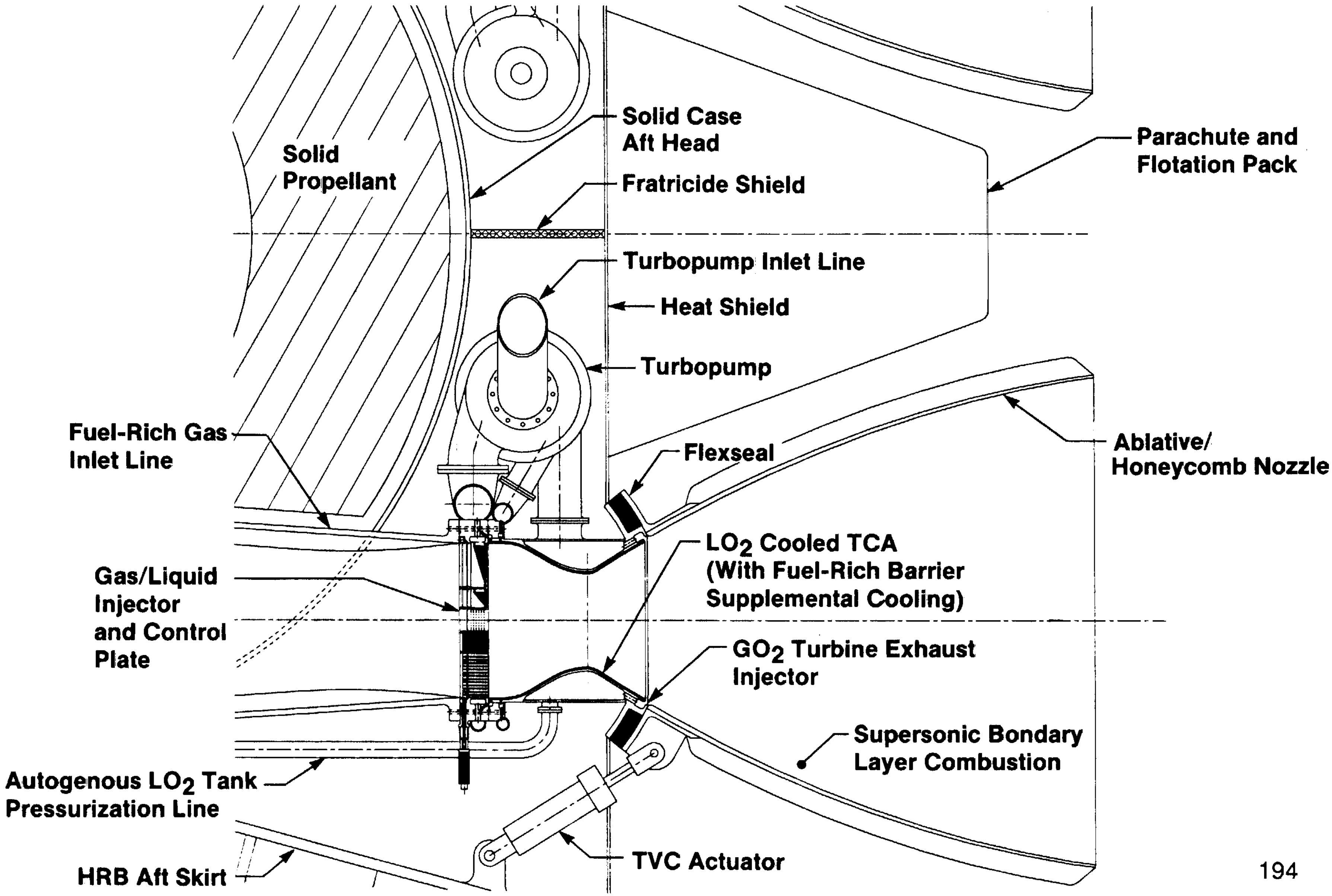
SLSC HRB ENGINE DETAILS



Our SLSC HRB Engine Detail Shows:

- A Warm, Fuel-Rich Gas Flow Control Plate in the Engine Inlet Duct, Whose Plan View Shape Is Identical to the Injector. It Can Be Rotated to Shut Off Gas Flow Area and It Preserves the Local Mixture Ratio Identical to the Overall MR Everywhere Across the Injector Face as It Moves. The Plate Is Supported by the Inlet Duct and Is Rotated by a Rack and Pinion Driven by a Torque Motor

- A Gas/Liquid, Vane Type LO_2/Warm Fuel-Rich Gas Injector With Two Sets of Involute Spiral Shaped Vanes and an Intermediate Hub. Trailing Edge Oxidizer Doublets Do Not Exist Near the Thrust Chamber Wall to Form a 1111°K (2,000°R) Gas Barrier There, Near the Circumferential O_2 Inlet Manifold

- A Supercritical O_2, Regeneratively Cooled Thrust Chamber With Two Pass Construction. O2 Supply Is From the Injector Manifold, and the Adjacent Outlet Manifold Supplies the Turbopump Assembly Turbine Inlet With 478 °K (860°R) Drive Fluid. The Throat of the Thrust Chamber Is Supported Against Nozzle Thrust and Gimbal Loads With a Thrust Collar That Doubles as a Turbine Exhaust Manifold

- A Silica Phenolic-Lined, Phenolic Honeycomb Nozzle Is Attached to the Thrust Chamber at Area Ratio 3.0 With an Annular Flexseal. The Turbine Exhaust GO_2 Flows Under the Flexseal From the Thrust Collar/Manifold and Injects Into the Boundary Layer of the Nozzle at Mach 1 Around the Regenerative Cooling Jacket Turnaround Manifold. High Velocity Ratio Shear Mixing Occurs Between the GO_2 and the Fuel-Rich Boundary Layer, Ignites, and Burns Off in the Supersonic Nozzle; Expanding to Area Ratio 26 to 1

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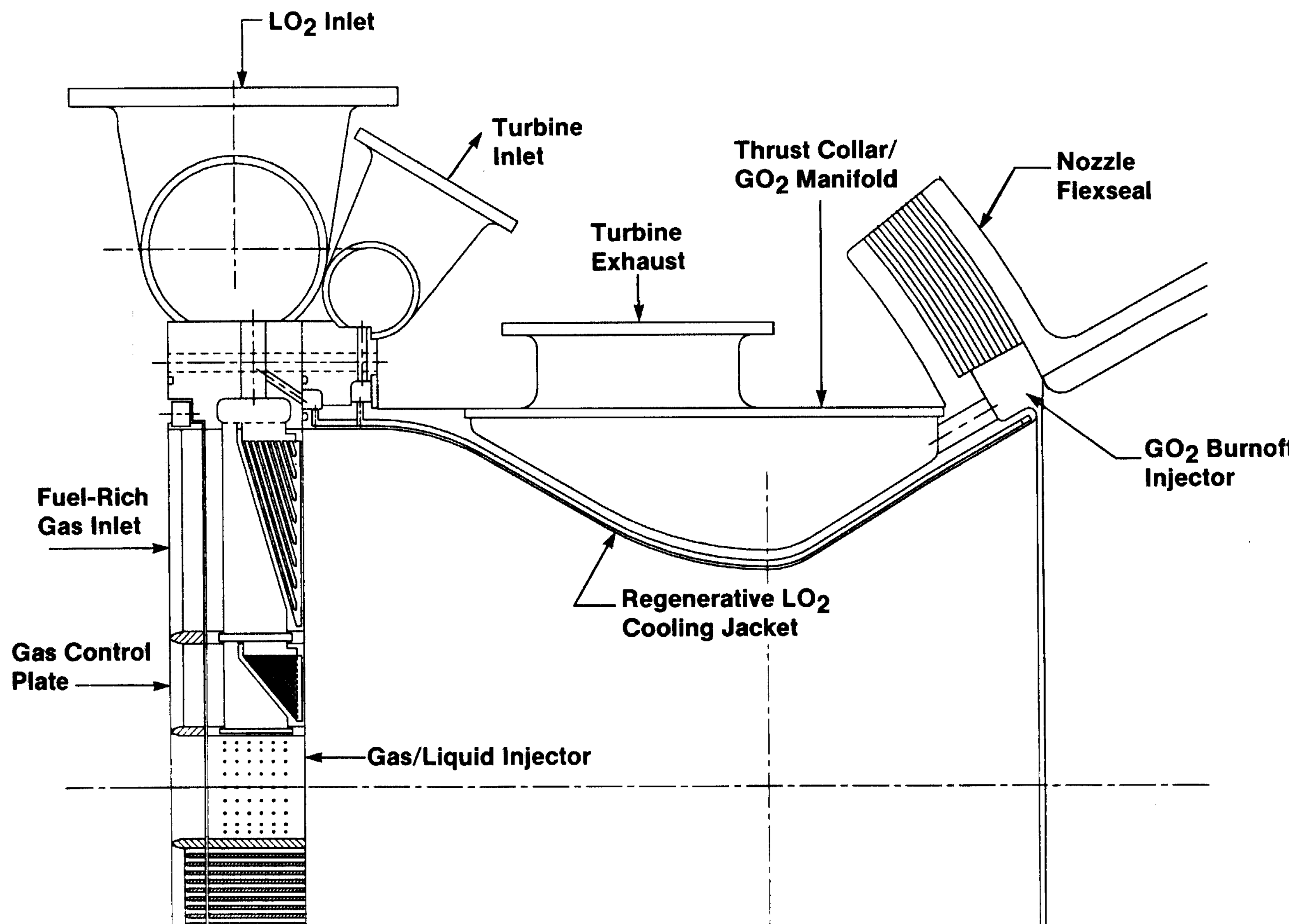
SLSC THRUST CHAMBER DETAILS



The Control Plate Has the Same Plan View Pattern as the Vaned Injector and Assembles Onto the Inlet Side of the Injector by Positioning in Between Three Locator Bearings and Attaching to the Actuator Arm. The Linear Solenoid Actuator Has a Bellows Seal Which Moves the Control Plate a Degree or So of Rotation to Change the Blockage Fraction of the Injector and Control Fuel-Rich Gas Flow

A Toroidal Feed Manifold Delivers LO_2 to the Outer Injector Vanes Through Between-Mount-Bolt Slots, a Concept Selected for Good Maintainability and Low Development Costs. Outer Vanes Inject LO_2 From Their Trailing Edges and Transport LO_2 to the Inner Vanes. Inner Vanes Do the Same, Providing LO_2 to the Inner Hub, Where Radial LO_2 Injection Provides Oxidizer to the Inner 2% of the Injector Area

All Outer Vanes Are Identical and All Inner Vanes Are Identical. Any Thermal Expansion Is Taken up by Elastic Vane Bending. Several Vanes and the Outer Hub May Be Extended Axially Into the Combustor to Form Baffle Pockets to Improve Combustion Stability, if Needed

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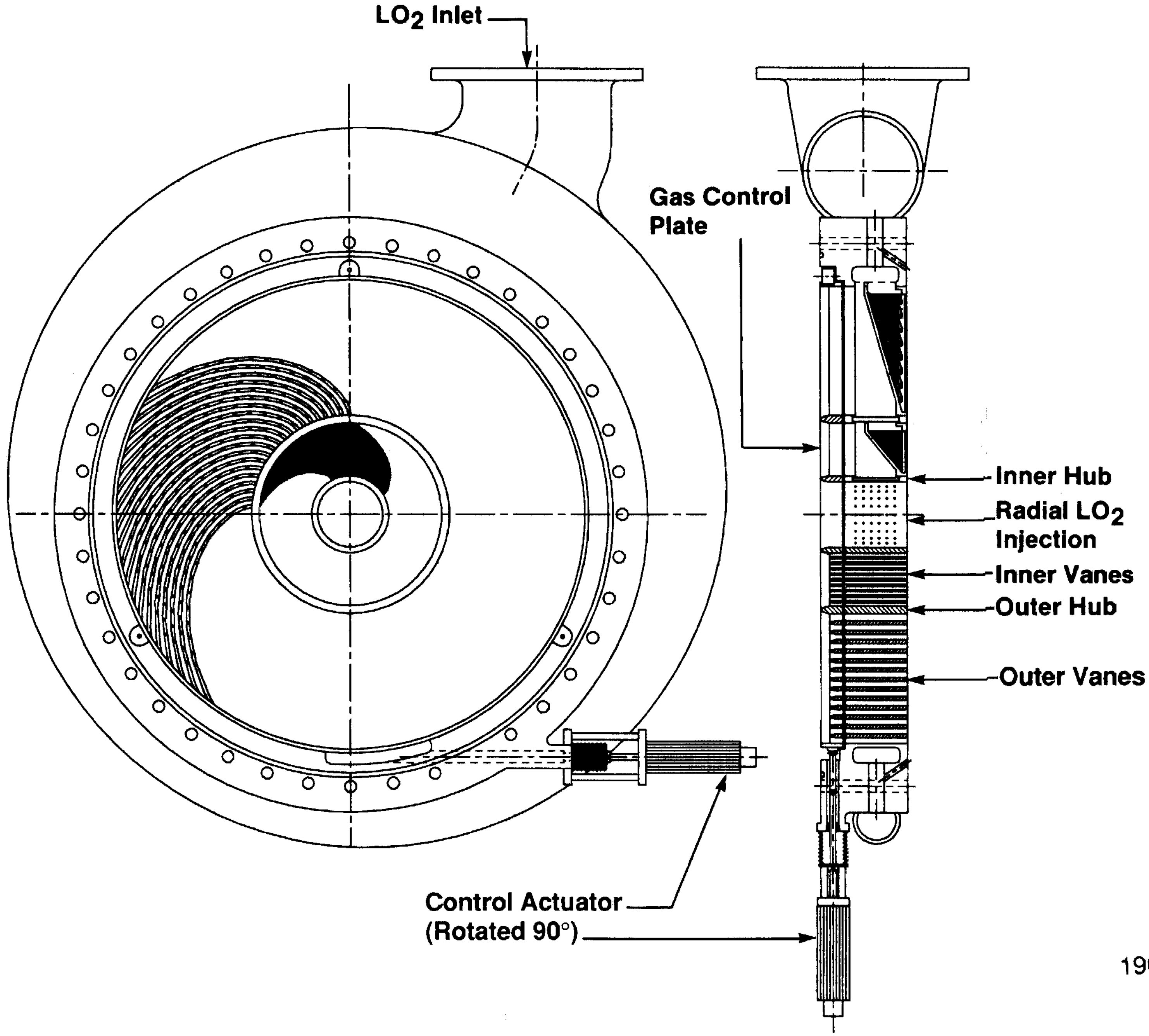
CONTROL PLATE AND ACTUATOR ASSEMBLY



LO_2 Injection Occurs From the Trailing Edges of the Vanes, the Balance of the Vane Acting as Manifold and Thermal/Structure. Like-on-Like Impinging Doublets Are Formed Beneath the Vane by Canting Orifice Pairs Toward Each Other Along the Vane Length. This Produces Spray Fans of Elliptical Cross Section With Major Axis at Right Angles to the Vanes. This Geometry Sprays Most Fluid Droplets Into the Fuel-Rich Gas Streams Adjacent to the Vane, Mixing the Propellants Prior to Completing Vaporization and Combustion

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Solid Propulsion

OUR INVOLUTE VANE INJECTOR IS THE BEST DESIGN CHOICE

Feature	Benefit
• Gas/Liquid Injector	• Best Combination of Combustion Performance and Stability
• Vaned Injector	• Proven Concept — Good Performance — Best for Axial Gas Inlet
• Involute Spiral Vane Shape	• Low Cost — All Vanes Alike (Two Places) — Fewer, Longer Vanes — Lowest Cost Control Plate • Stability — Hub and Some Spiral Vanes Can Become Baffles by Lengthening Them Axially • Performance — Uniform Vane Spacing Everywhere — Constant MR Distribution During Control Plate Rotation • Thermal Expansion — Curved Vane Allows Thermal Growth to Be Taken in Elastic Bending

The Combustor Will Be Conical With a Contraction Ratio of Three, Giving a 76.2 cm (30-in.)Diameter Injector. 63.5 cm (25-in.) Combustor Length Is Estimated to Be Adequate for High Performance With a Vaned Injector and Fuel-Rich Gas of Proper Design and Composition

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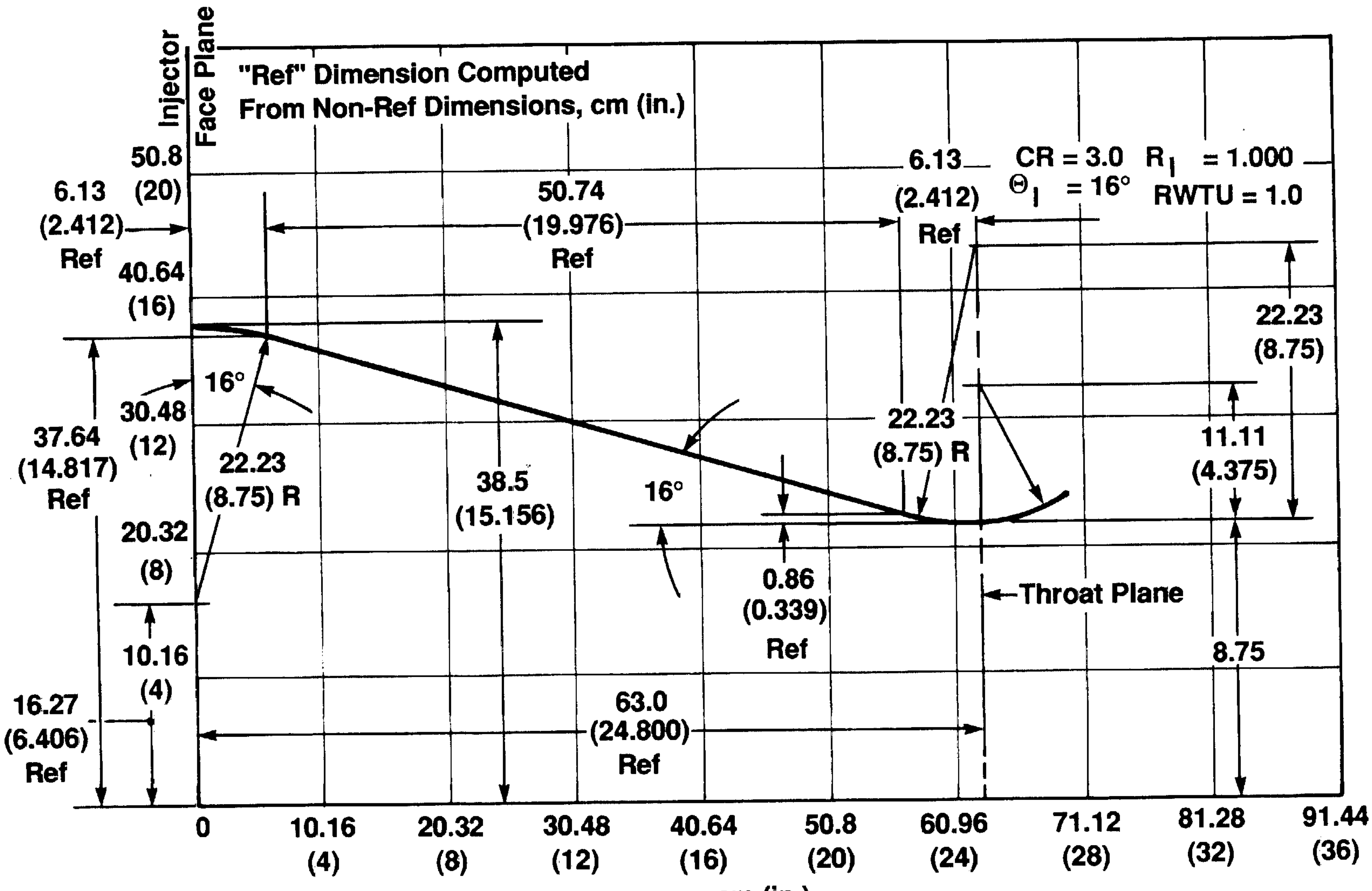
3.56 MN (800 LBF) TAPERED CHAMBER AND NOZZLE GEOMETRY*



*12.4 MPa (1,800 psia) Design

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OUR BURNOFF NOZZLE IS THE BEST DESIGN CHOICE

Features	Benefits
• Burns Off Fuel-Rich TCA Barrier Coolant With GO_2 TPA Drive Gas (Uses High Velocity Ratio Shear Mixing and Supersonic Combustion)	• Increases Engine I_{sp} • Cools Ablative Nozzle at a Low Area Ratio (~3 to 1) • Reduces TCA Wall Cooling Requirements • Eliminates TPA Seals
• Simplifies Supersonic Splitline TVC Nozzle Interface	• Provides Boundary Layer Gas Flow for Splitline Joint • Eliminates Seals (Except Flexseal)

Our LO_2 Turbopump Is Compact, Having a Single-Staged Pump and Turbine. Its Conventional Bearings Are LO_2 Lubricated and Cooled From the Pump; Controlled O_2 Leakage Is Routed to the GO_2-Driven Turbine. There Are No Buffer Nor Low Leakage Seal Packs in the Design. TPA Specifications Are Given on the Following 2 Pages

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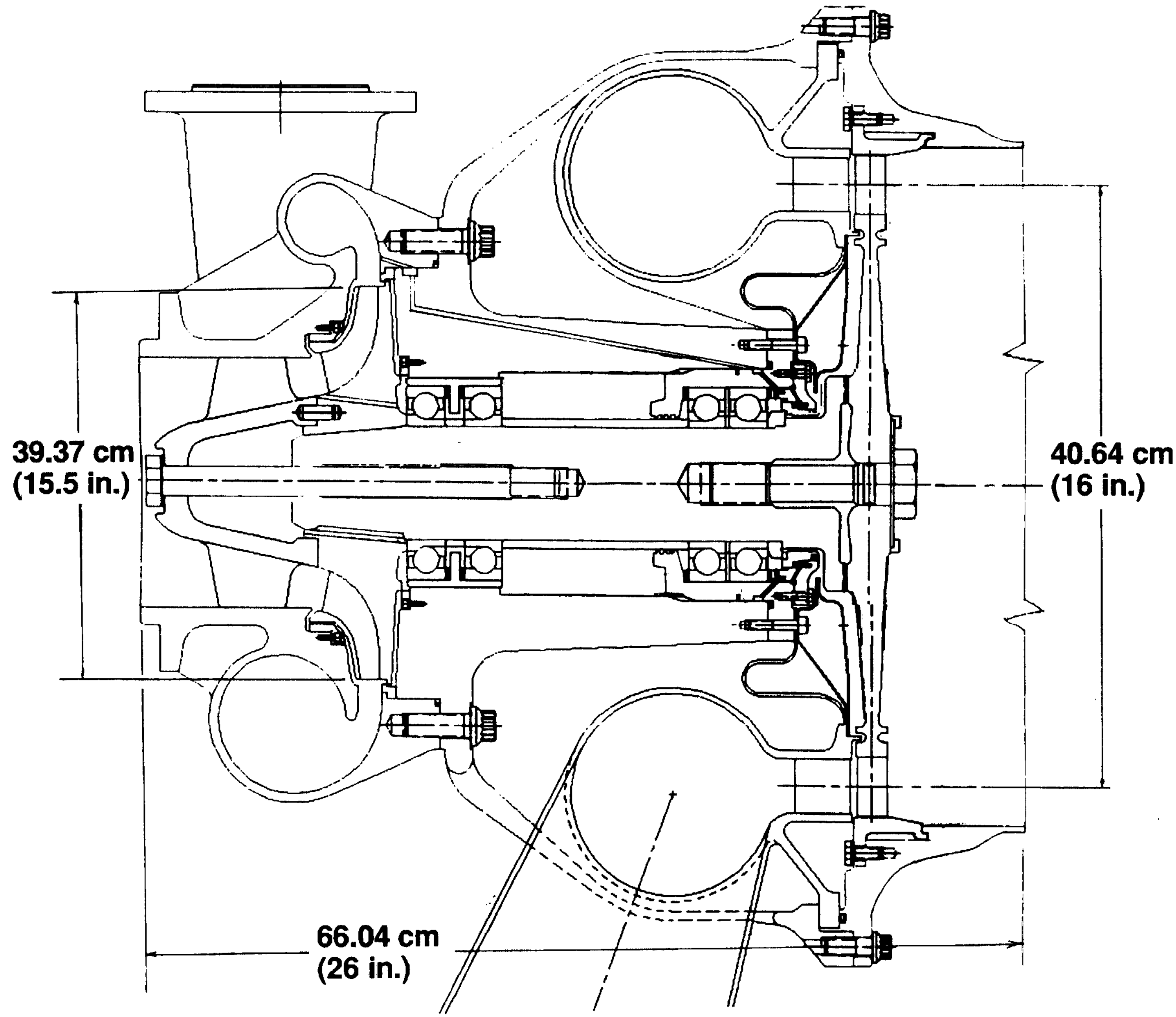
OUR HRB TURBOPUMP IS SIMPLE - IT HAS NO SEALS



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TURBOPUMP DESIGN SPECIFICATION SUMMARY

Pump Design Parameters

	Units	4 TPAs
Number of Stages		1
Suction Pressure	kPa (psia)	413.7 (60)
Inlet Temperature	°K (°F)	91 (-297)
Disch. Pressure	MPa (psia)	14.1 (2,040)
Weight Flow	kg/sec (lbm/sec)	764.9 (1,686.26)
Flow Rate	m^3/min (gpm)	40.23 (10,629)
Tip Diameter	cm (Inch)	39.32 (15.48)
Inducer Diameter	cm (Inch)	29.31 (11.54)
Blade Exit Width	cm (Inch)	3.38 (1.33)
Volute Area	cm^2 ($in.^2$)	115.68 (17.93)
Pump Efficiency	--	0.809
Specific Speed	rpm * $gpm^{.5}/ft^{.75}$	1,580
Suction Specific Speed	rpm * $gpm^{.5}/ft^{.75}$	27,000
Shaft Speed	rpm	7,716
Shaft Power	kW (hp)	11 318.1 (15,177.8)
Torque	J (ft-lbf)	14 007.2 (10,331.2)

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TURBOPUMP DESIGN SPECIFICATION SUMMARY

Tabulation of HRB Turbine Study

	Units	4 TPAs
Turbine Type		Impulse
% Admission		100.0%
Number of Stages		1
Number of Turbines		4
Power/Turb	kW (hp)	11 318.1 (15,177.8)
Speed	rpm	7,716
pti	MPa (psia)	11.31 (1,640)
tti	°K (°R)	476 (856)
pte	kPa (psia)	1 723.7 (250)
PR		6.56
Flow Rate With Turb	kg/sec (lbm/sec)	90.83 (200.25)
Mean Diameter	cm (in.)	40.64 (16)
Hub/Tip Ratio	Degrees	0.926
Rotor Turn	Degrees	116.61
u/Co	cm/sec (ft/sec)	7.65 (0.251)
u	m/sec (ft/sec)	164.2 (538.7)
Hp/Flow Rate	kW/kg/sec (hp/lbm/sec)	124.6 (75.79426)
Turbine Efficiency	--	0.584
Total Flow Rate	kg/sec (lbm/sec)	90.83 (200.25)
Power Total	kW (hp)	11 318.1 (15,177.8)
Gamma	kg/mole (lbm/mole)	0.641 (1.413)
MW	kg/mole (lbm/mole)	14.5 (32)
R	J/kg°K (ft-lbf/lbm-Degree R)	2 663.3 (49.5000)
Cp	kJ/kg°K (Btu/lbm-Degree R)	1.06 (0.2537)
Bearing ID	mm	80-130
Maximum Bearing DN		1.00E + 06

8.6.6

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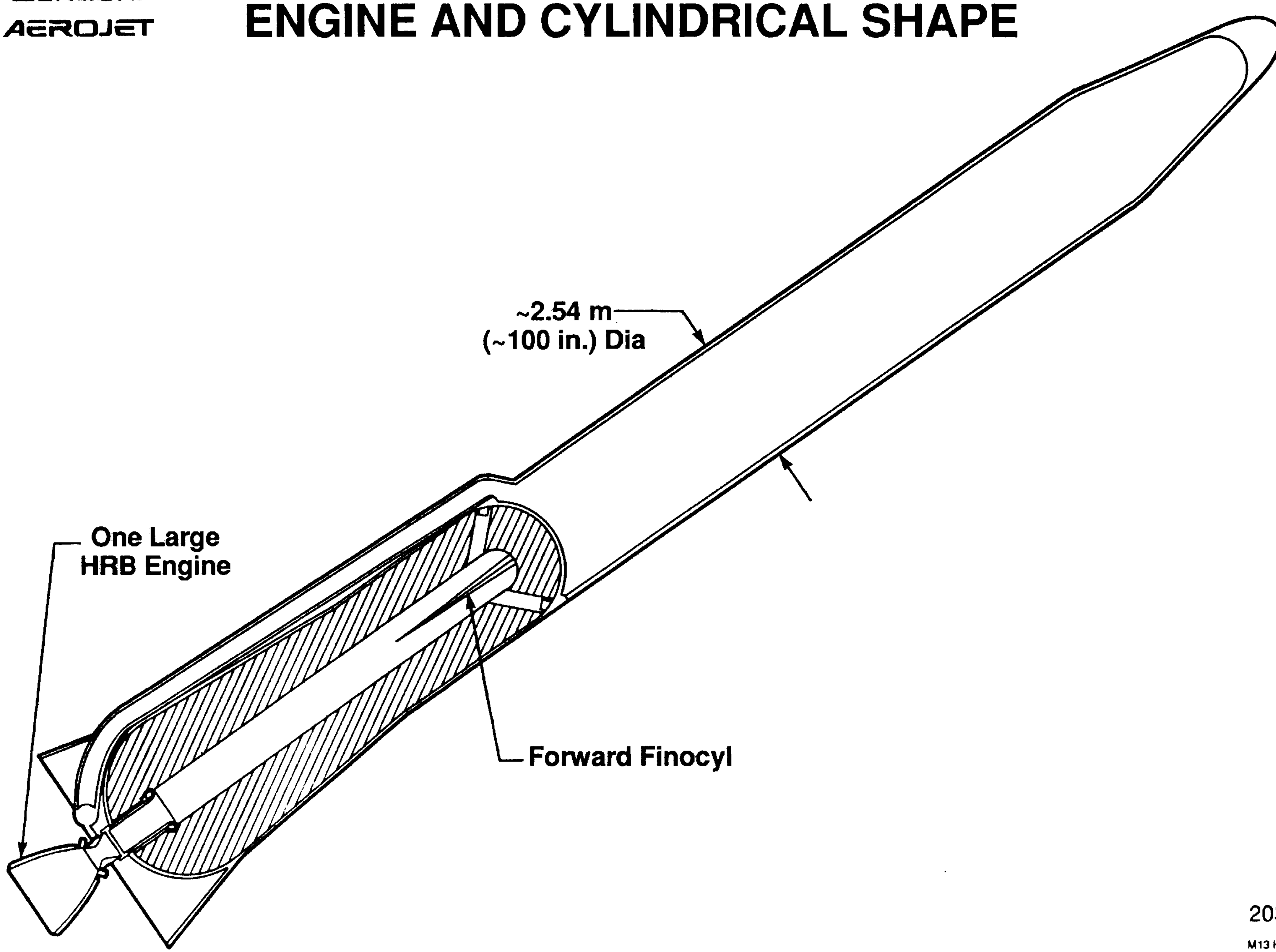
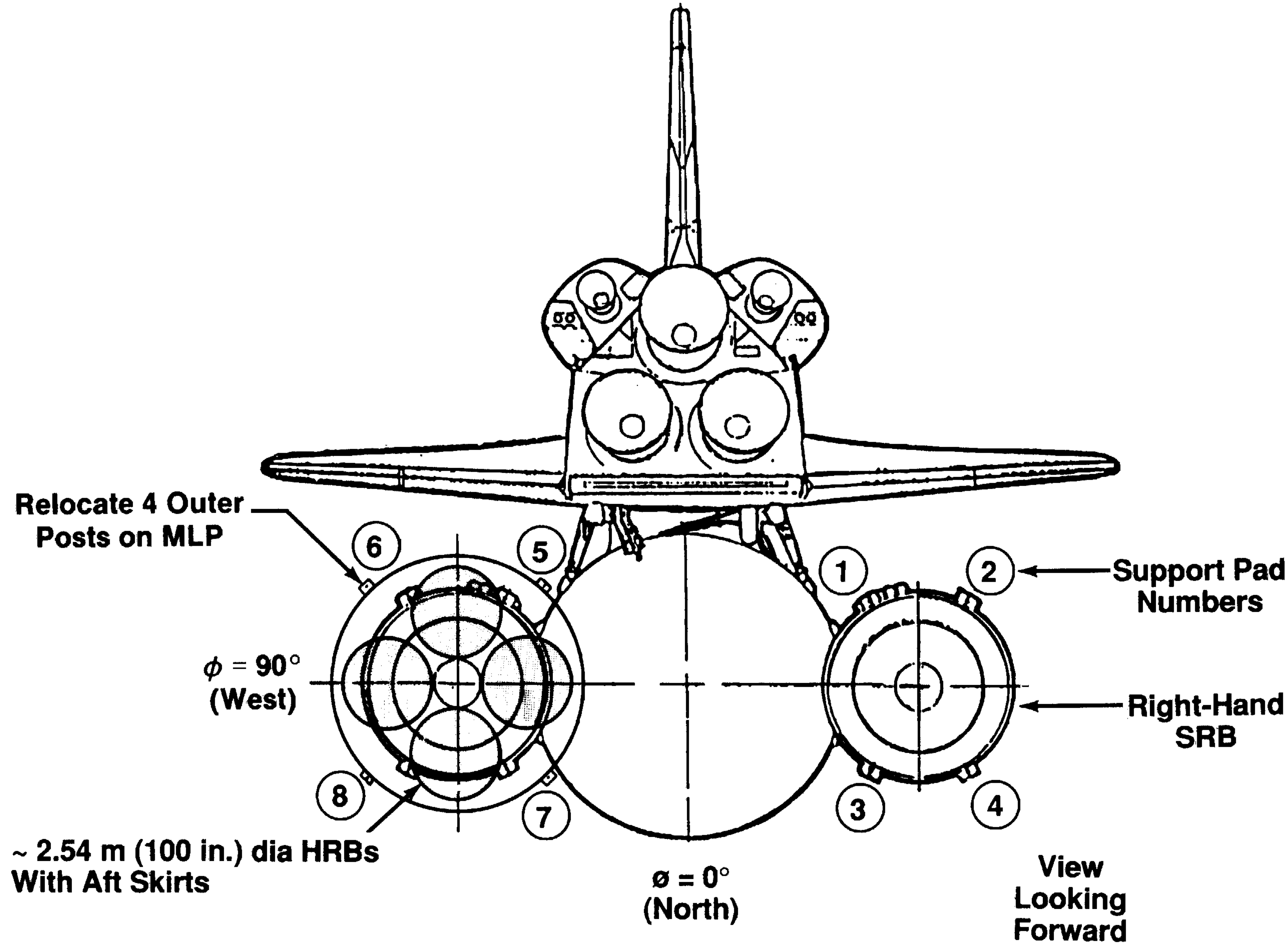
OUR SMALL SLSC HRB HAS A SINGLE ENGINE AND CYLINDRICAL SHAPE



LARGE OR SMALL HRB GEOMETRY REQUIRE MINIMAL MLP INTERFACE CHANGES



DESIGN ANALYSES

Our SLSC Engines Can Be Thrust Throttled by Controlling the Oxygen Flow to the Injector. A Turbine Bypass Valve Is Used to Control the Power to the TPA Without Starving the Regeneratively-Cooled Thrust Chamber for Coolant. When LO_2 Flow Is Decreased, the Thrust Chamber Pressure Drops With the Thrust Level, and This Is Reflected in a Reduced Solid Case Pressure. Because the Solid Grain Burn Rate Varies With Approximately the Square Root of the Case Pressure, a Mixture Ratio Shift Will Occur During Throttling. Throttling Down From the Design Point Causes a Fuel-Rich MR Shift, as Shown on the Left Hand Side of the Curve Below. It Shows, for Example, That Reducing the LO_2 Flow to 2/3 of the Nominal Design Value Will Cause the Vacuum Thrust to Fall to About 3/4 of Nominal Thrust and the MR to About 0.83 (or 5/6) of Nominal MR. In Our Case the MR Would Fall From 2.6 to 2.15, and About 3.5 sec of Specific Impulse Is Lost. Conversely, Throttling up LO_2 Flow by 1/3 Increases Thrust by About 1/5 and MR by 1/6. This Loses About 6 sec I_{sv}; However, Our Design Does Not Include Throttling up, but Only Down

A Control Plate at the Injector Inlet Can Be Opened Somewhat When Throttling Down to Reduce the Case Pressure and Burning Rate to Reduce MR Shift and Improve System Performance Through I_{sp} Control and Propellant Utilization

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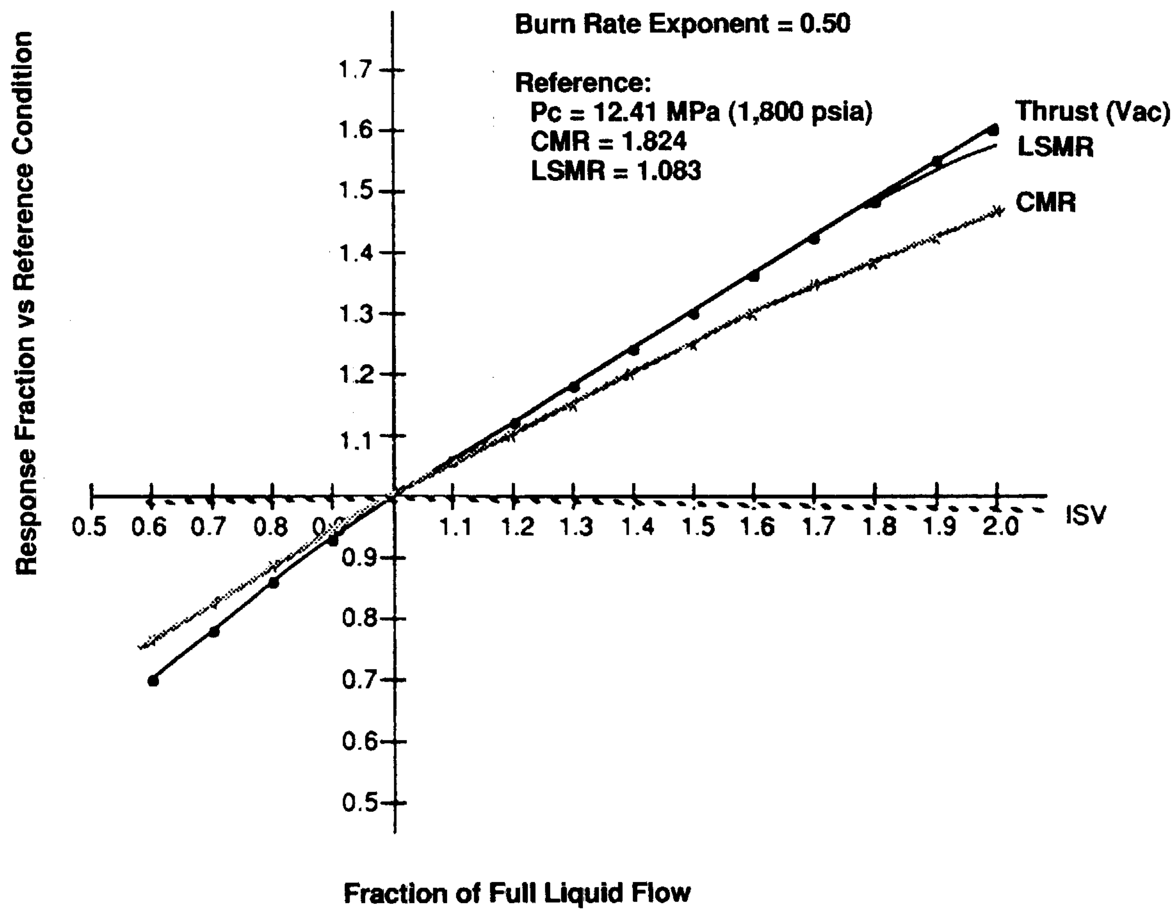
SLSC CONTROL WITH LO_2 AND SOLID PROPELLANT NO. 7



M4D10/Pt 2

Control Plates Have Been Included in Our Design, as New Technology. They Are Developable Items Not Requiring a Breakthrough and Can Perform Many Important Functions in Addition to the Specific Impulse and Propellant Utilization Control Previously Mentioned

The Warm Gas Flow Control Plates Can Be Used to Trim Relative Flow to the Four Engines From the Single Solid Propellant Gas Generator. The Injector ΔP Control Should Prove Useful During System Development in Reducing Hardware Iterations

During Flight, an Engine Failure Will Result in Termination of LO_2 Flow. The Control Plate, When Closed to Its Minimum Flow Area, Will Control Warm Gas Flow to Its Nominal, Operational Value, Permitting the Other Three Engines to Operate to a Safe Abort (or, Possibly, to Mission Completion). If All Four Control Plates Are Fully Opened When All LO_2 Flow Is Stopped, Solid Case Extinguishment Is Assured.

Partially Closed Control Plates May Aid Ignition and Speed the Start Transient. The Uncooled Control Plate May Also Be Used to Protect the Leading Edge of the Cryogenically Cooled Injector Vane From Thermal Shock and From Solid and Liquid Particle Impingement

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WE HAVE GAS FLOW CONTROL PLATES FOR MANY REASONS

1. Uniform Gas Flow Across Injector Inlet and Between TCAs
   - Uniform Gas Flow Reduces MRD Losses in Combustor
   - Same Flow to Each TCA
   - Reduces Development Time and Cost
2. Gas Flow Control to Failed/Shut Down TCA
   - Allows Safe Abort Operation of Other Three TCAs by Controlling Gas Flow to Operational Rate
   - Assures Extinguishment for Launch Pad Abort
3. Independent Thrust and MR Control for HRB
   - Trims Motor Case Pressure Independently of Combustor Pressure
4. Reduces Gas Flow During Ignition Transient
5. Protects Injector Inlet From Solid Particles and Heat

Injector Gas Velocity Controls Injector Performance in Areas of Specific Impulse, Pressure Schedule, Combustion Stability, and, Potentially in Mechanical Ways, Flutter, Heat Transfer, and Longevity. The Chart Below Shows That for a Given Gas Flow Rate, Velocity Is Controlled by Injector Size, or Contraction Ratio (Area vs Throat Area), and Percent Blockage Caused by the Injector Structure and Manifolding. It Shows, for Example, That With 50% Blockage the Gas Velocity Will Be About 55 m/sec (180 ft/sec) for an HRB Engine of 3.6 MN (800,000 lbf)Thrust Class and a Contraction Ratio of 3.0. A 0% Blockage Is Impossible, Even With a Radial Injector, Because the Radial Liquid Streams That Penetrate the Gas Flow With That Concept Block Gas Flow Nearly as Do Our More Controllable Injector Vanes. Thus, Gas Injection Velocities Will Be Greater Than 30.5 m/sec (100 ft/sec)

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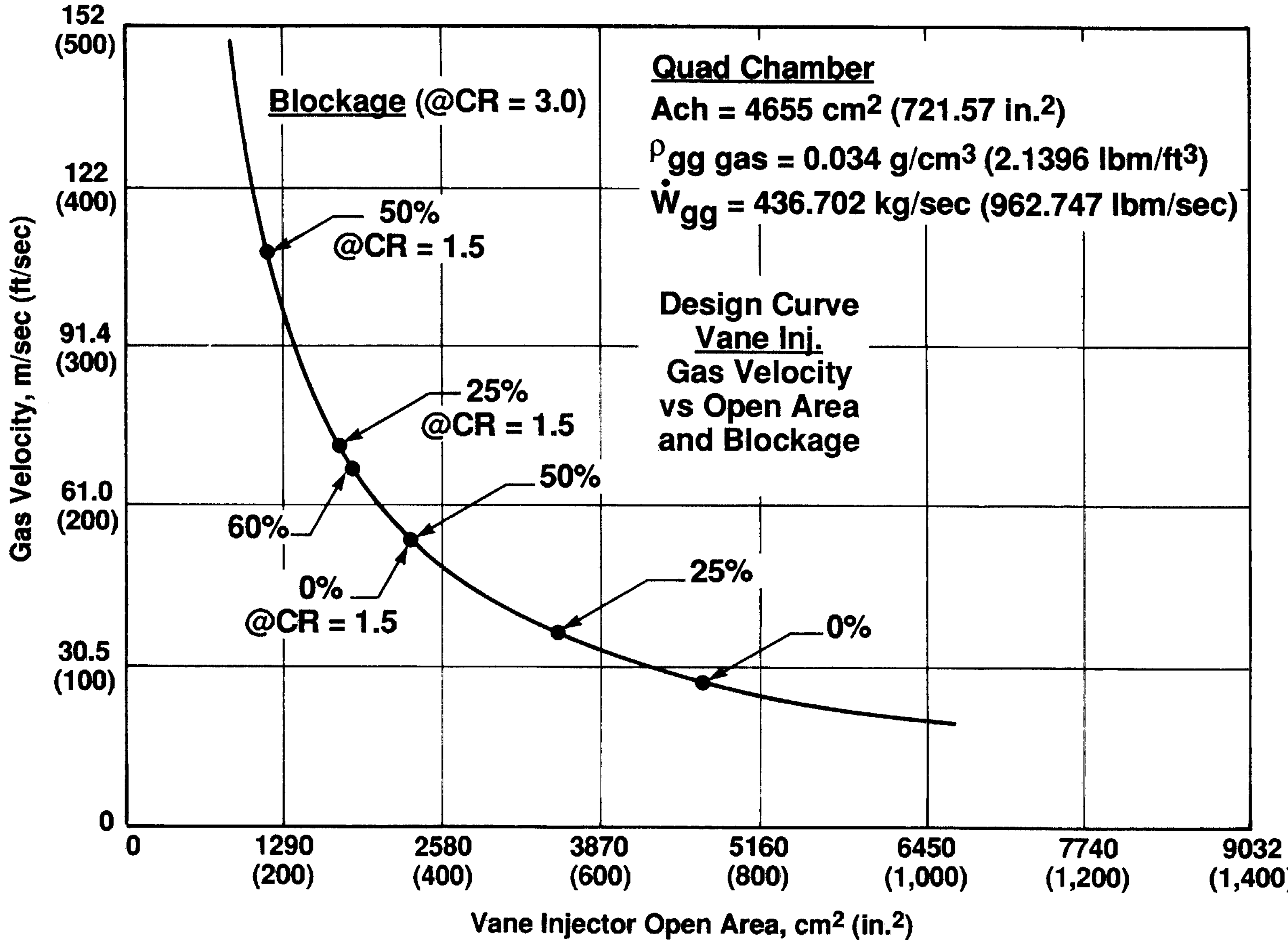
INJECTOR GAS VELOCITY DEPENDS ON COMBUSTOR CONTRACTION RATIO AND BLOCKAGE PERCENT



Gas Injection Velocities of 30.5 to 61.0 m/sec (100 to 200 ft/sec) Suggest That Radial Injectors Cannot Be Used for High Performance, HRB Systems. Our Injector Diameter (CR = 3.0) Is Approximately 76 cm (30 in.) (for One of Four Engines/HRB), and the Chart Below Shows That 0.95 to 1.60 cm (3/8 to 5/8-in.)-Dia LO_2 Orifices Would Be Required to Penetrate to the Center of the Gas Flow When Injected From the Periphery (Radial Injectors Have No Hardware in the Gas Flow Stream). Poor Local O/F Mixture Ratio Control and Low Performance Would Result

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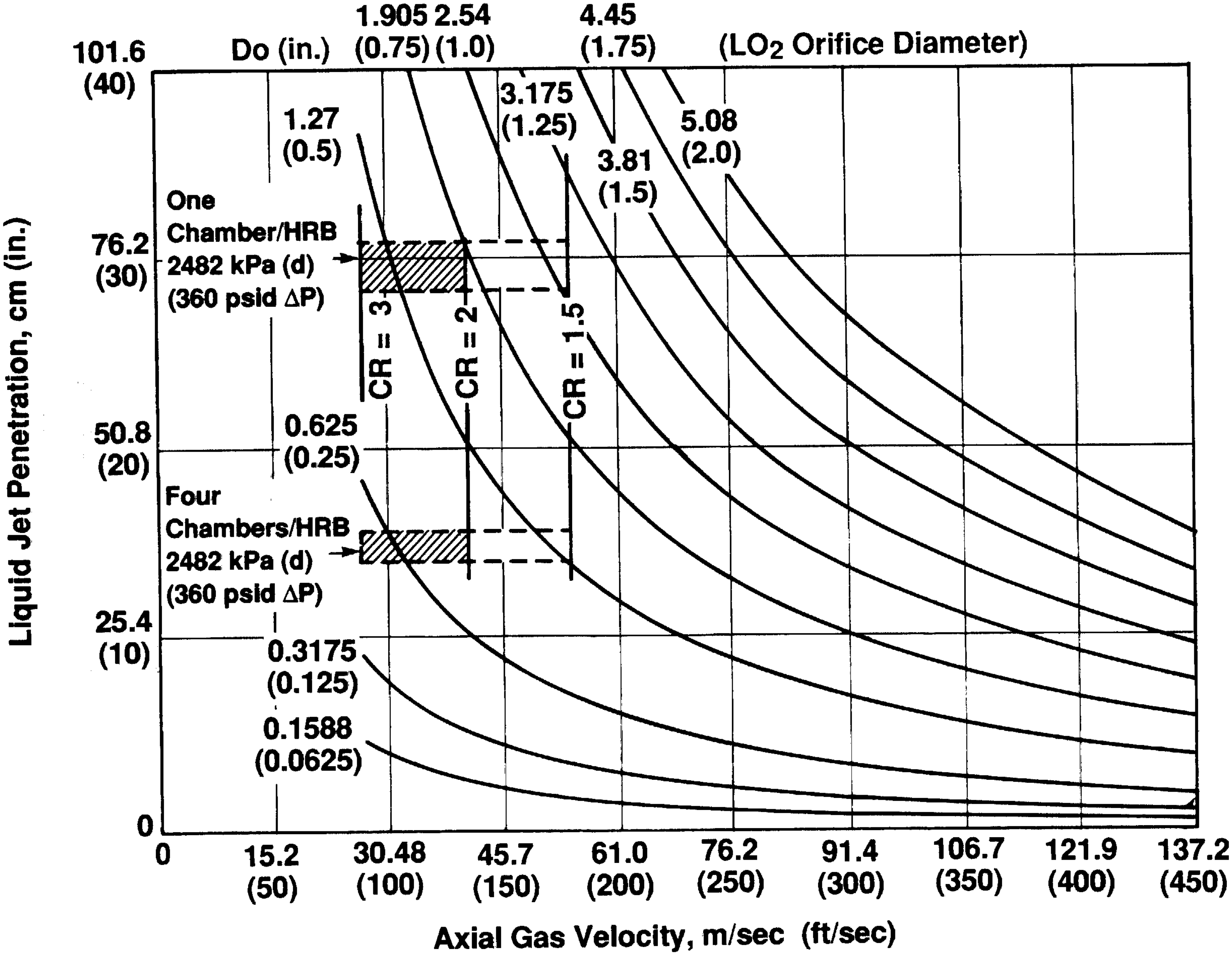
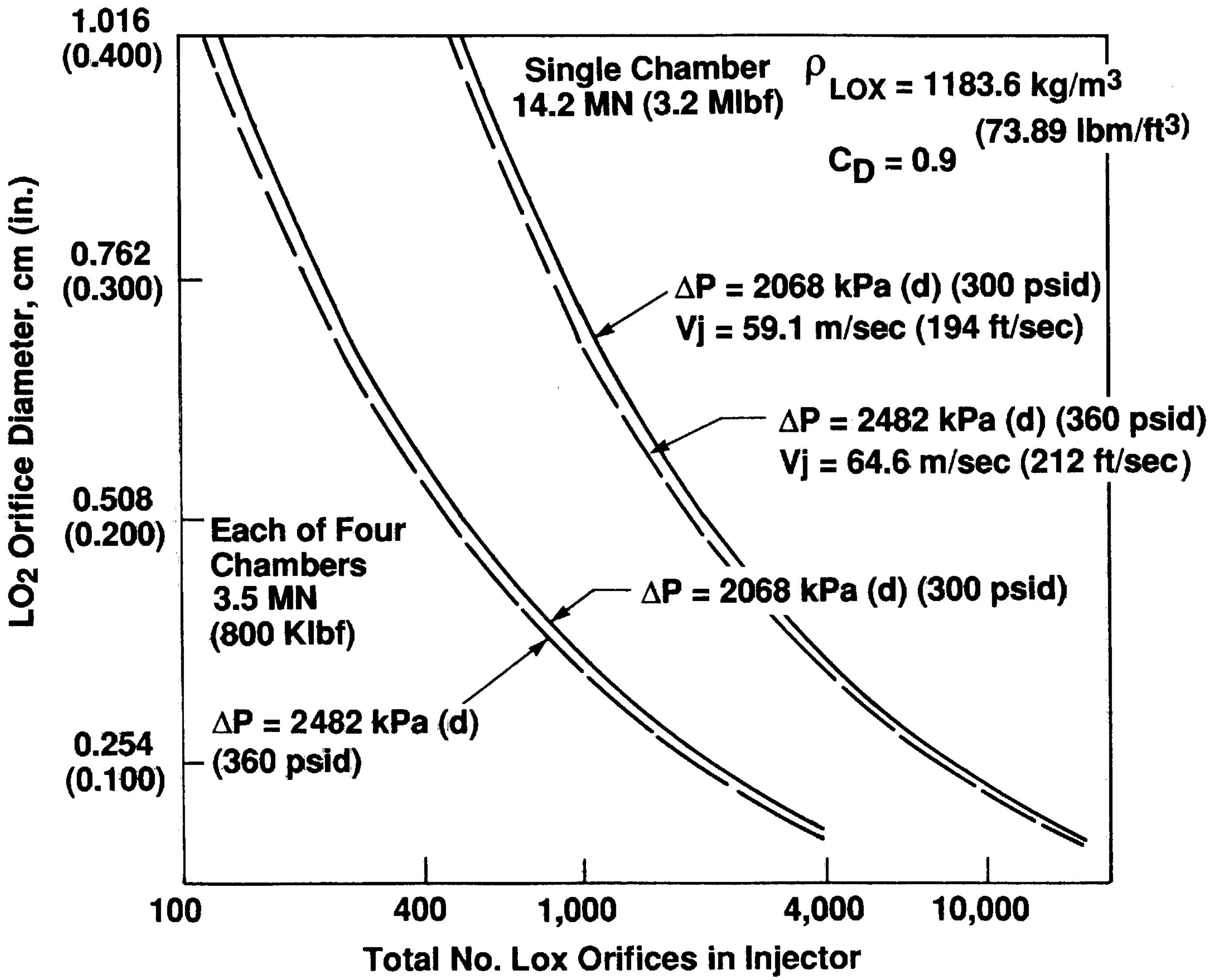
ADEQUATE PENETRATION IN A MONOLITHIC-SIZED RADIAL-INJECTION COMBUSTOR REQUIRES LARGE INJECTOR ORIFICES AND LEADS TO POOR ERE*



Vaned Injectors Can Use Reasonable Numbers of Reasonable Sized LO_2 Orifices to Provide Good Engine Operation. For Example, 0.25 cm (0.10-in.) Dia Injection Orifices Result From a Design Having About 2,500 Orifices. This Could Be Eight Doublets on the Trailing Edge of 80 Vanes, With 80 Identical Outer Sections and 50 Identical Smaller Inner Sections as Shown on Page 195

GENCORP AEROJET

VANED INJECTORS CAN USE SMALLER INJECTOR ORIFICES AND GIVE HIGHER ERE*



*Energy Release Efficiency

Fuel Film Cooling to Augment TCA Primary Cooling (Ablative or Regenerative) Is Achieved Easily in Our HRB Engine by Injecting LO_2 Near the Combustor Wall. There Are Two Difficulties, However. One Is That Use of Excessive Amounts Reduce Engine Performance Significantly, Under Normal Conditions. These Losses Are Reduced in Our Design, Because the Fuel-Rich Boundary Layer Is Burnt Off in the Supersonic Nozzle With GO_2 Turbine Exhaust

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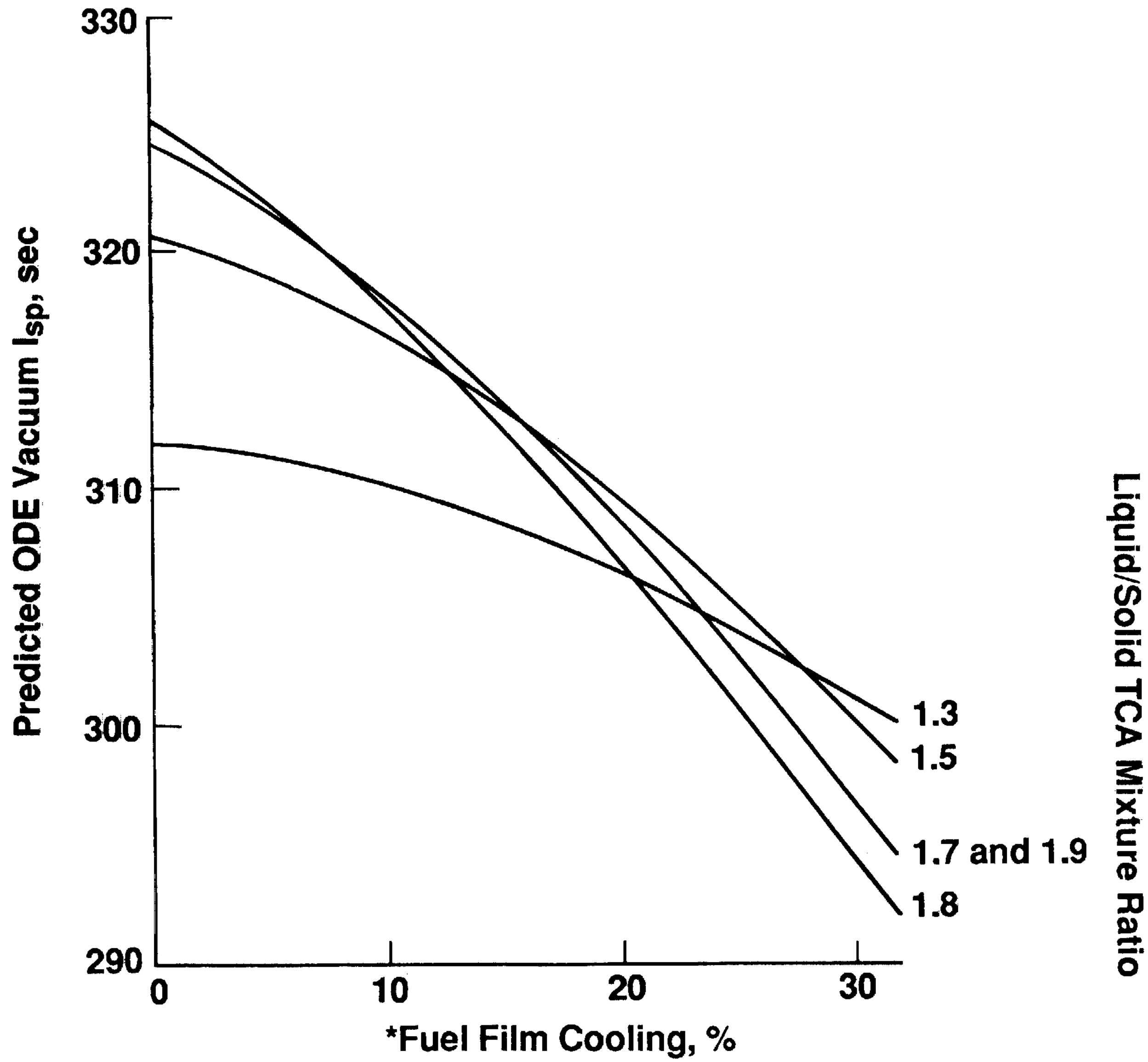
OUR SLSC HRB PERFORMANCE IS BOUNDED MR AND % FFC*



Another, Closely Related Factor Is That the Effectiveness Is Limited by the Time the Barrier Reaches the Throat Plane, Where It Is Needed the Most. This Suggests That a Portion of the FFC Should Be Introduced Just Upstream of the Throat to Improve Throat Cooling and Engine Performance

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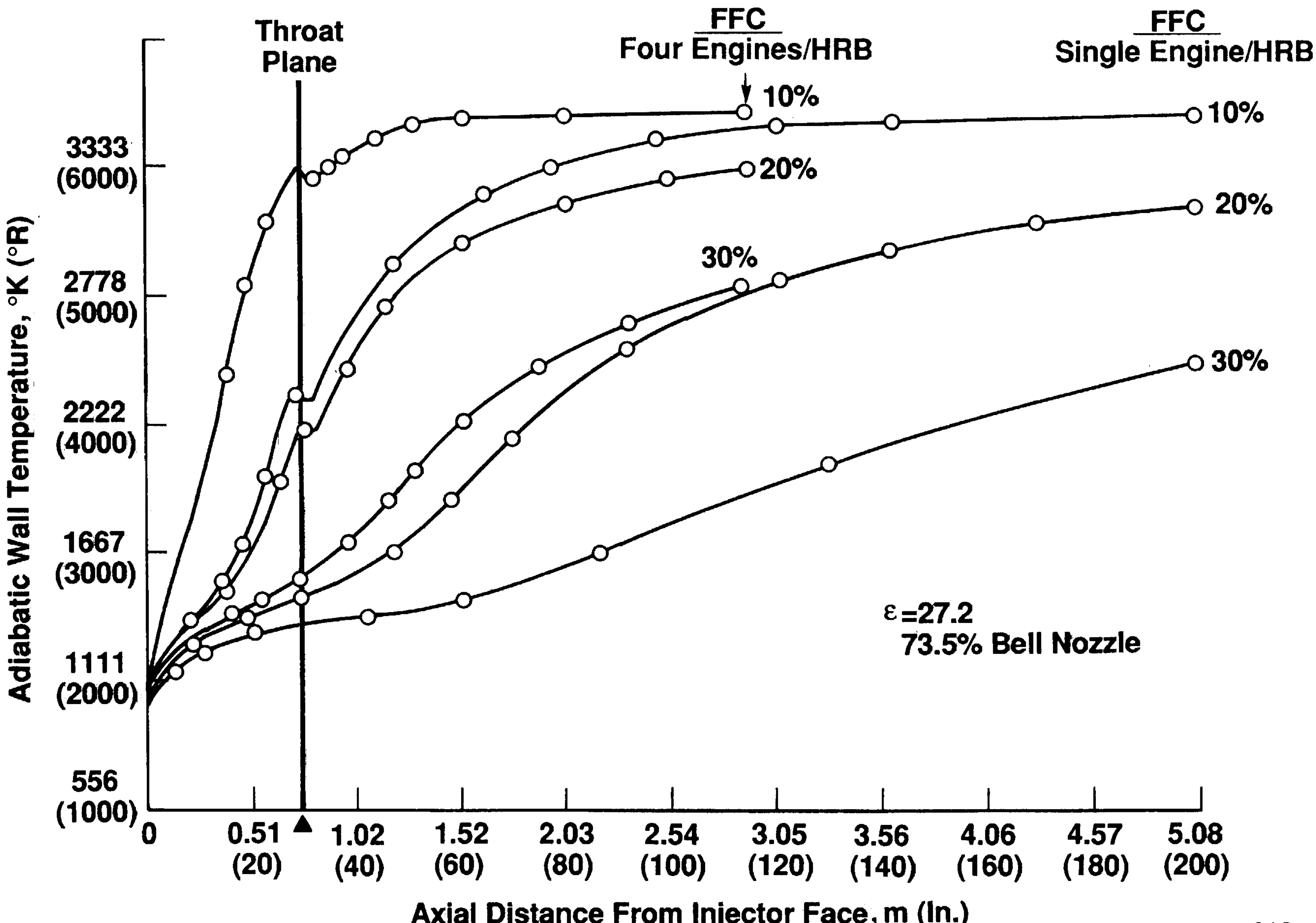
RISING ADIABATIC WALL TEMPERATURE REFLECTS ENTRAINMENT OF HOT CORE INTO THE FFC BARRIER



Analysis Shows That Silica Phenolic Throats Require About 15% Fuel Film Cooling. Alternatively, We Learned That About the Same Amount of Coolant Will Supplement a LO_2 Regeneratively Cooled TCA Wall and Provide Turbine Drive Fluid as Well. Either Approach Avoids Use of the Currently Expensive, Large Sized Carbon-Carbon Throat Inserts That Requires Little Coolant

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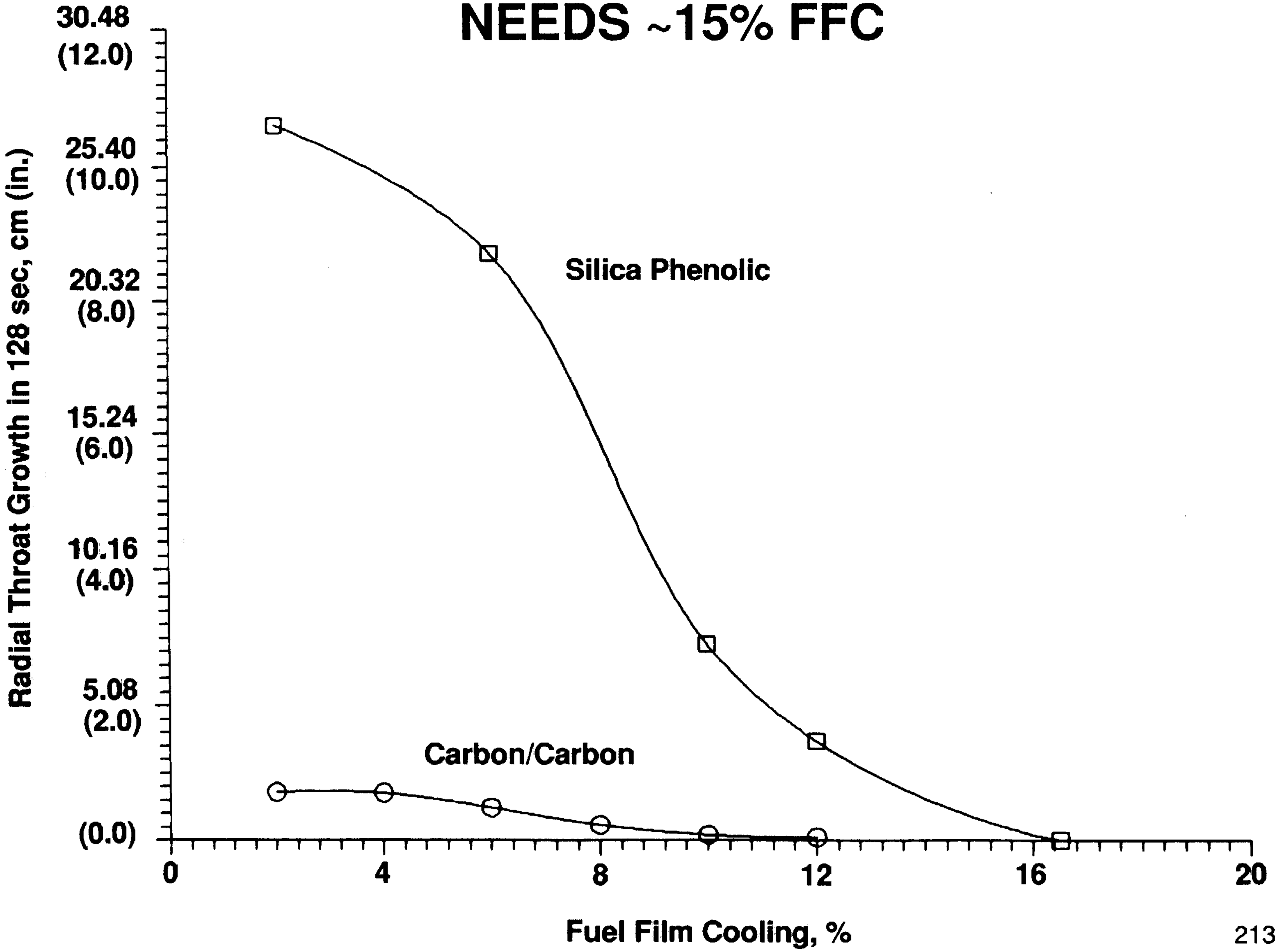
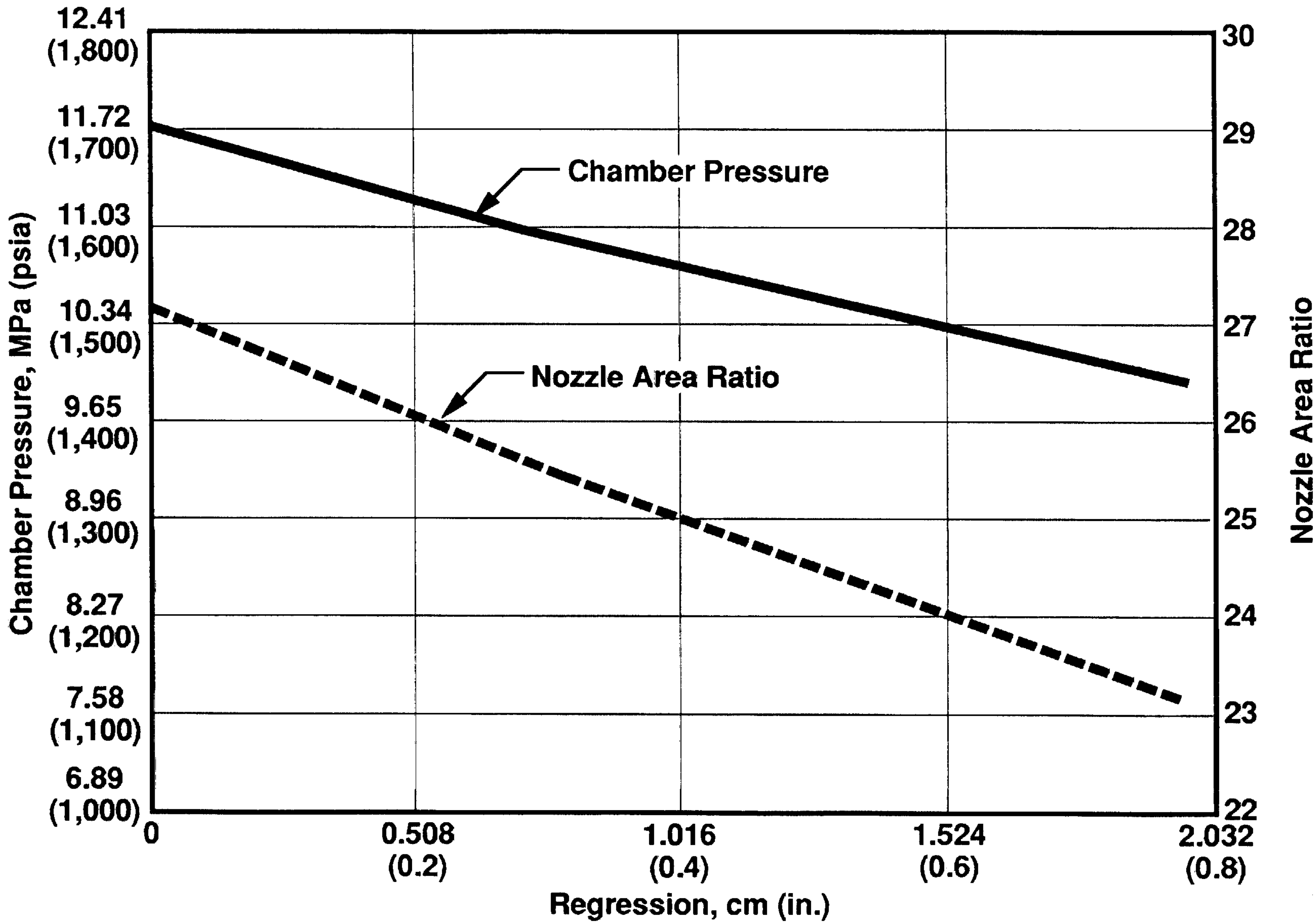
HRB THROAT RECESSION ANALYSIS SHOWS SILICA PHENOLIC NEEDS ~15% FFC



Failure to Protect Ablative Throat Sections Results in Throat Area Growth During Operation, Which Reduces Chamber Pressure and Nozzle Area Ratio, and, Hence, Engine Specific Impulse **(See Page 215)**

THROAT REGRESSION REDUCES TCA PRESSURE AND NOZZLE AREA RATIO

Chamber Pressure and Area Ratio vs. Regression



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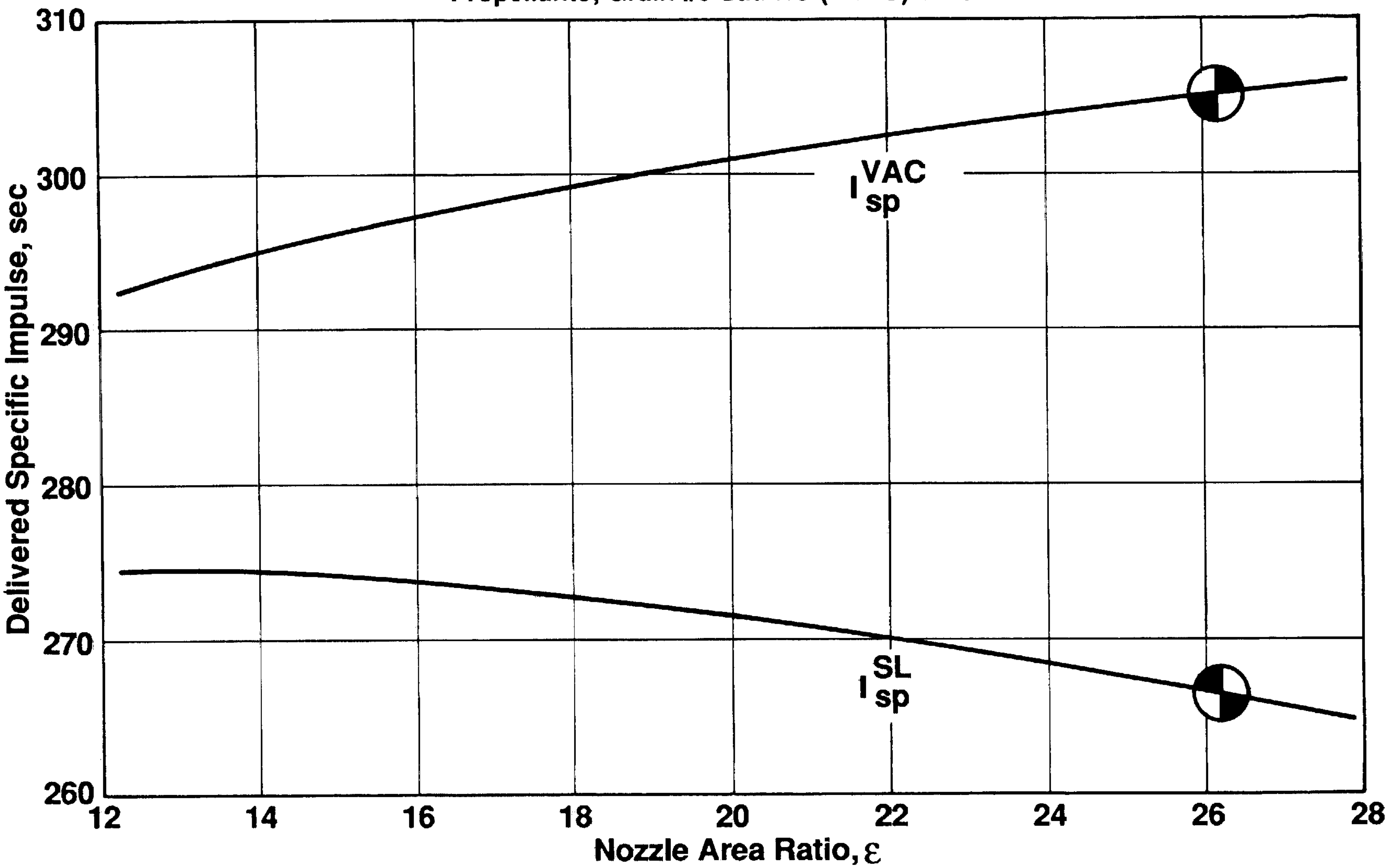
REDUCTION IN AREA RATIO LOWERS VACUUM SPECIFIC IMPULSE

Turbopump Fed (TF) HRB from Preliminary Design Case; 8 D EBBC - 1, PC = 11.72 MPa (1,700.00 psia) Propellants; Grain #8 Sat. HC (PEBC) + LO2



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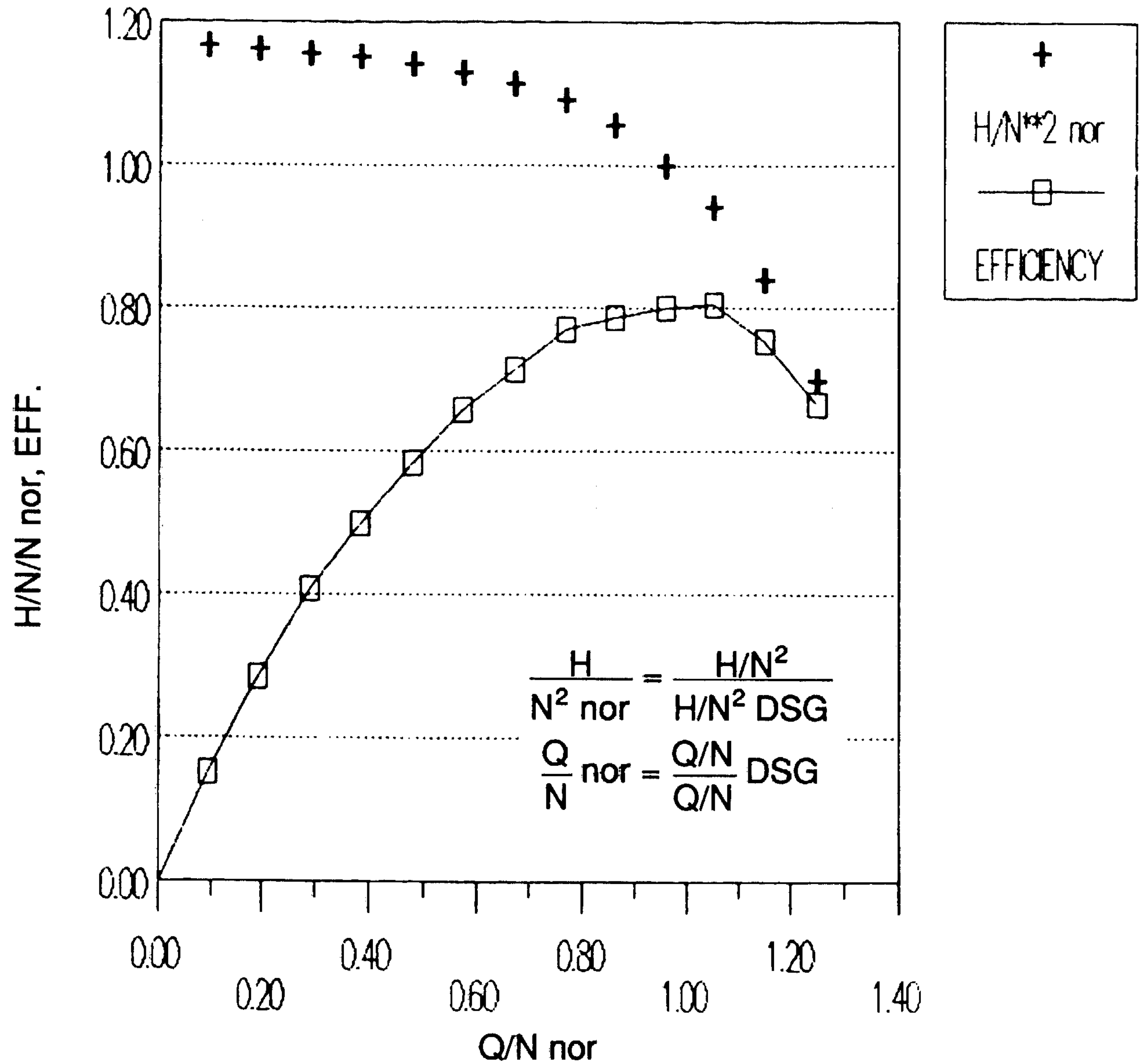
OUR HRB PUMP PERFORMANCE IS CHARACTERIZED



$$\frac{H}{N^2}\text{ nor} = \frac{H/N^2}{H/N^2\text{ DSG}}$$

$$\frac{Q}{N}\text{ nor} = \frac{Q/N\text{ DSG}}{Q/N}$$

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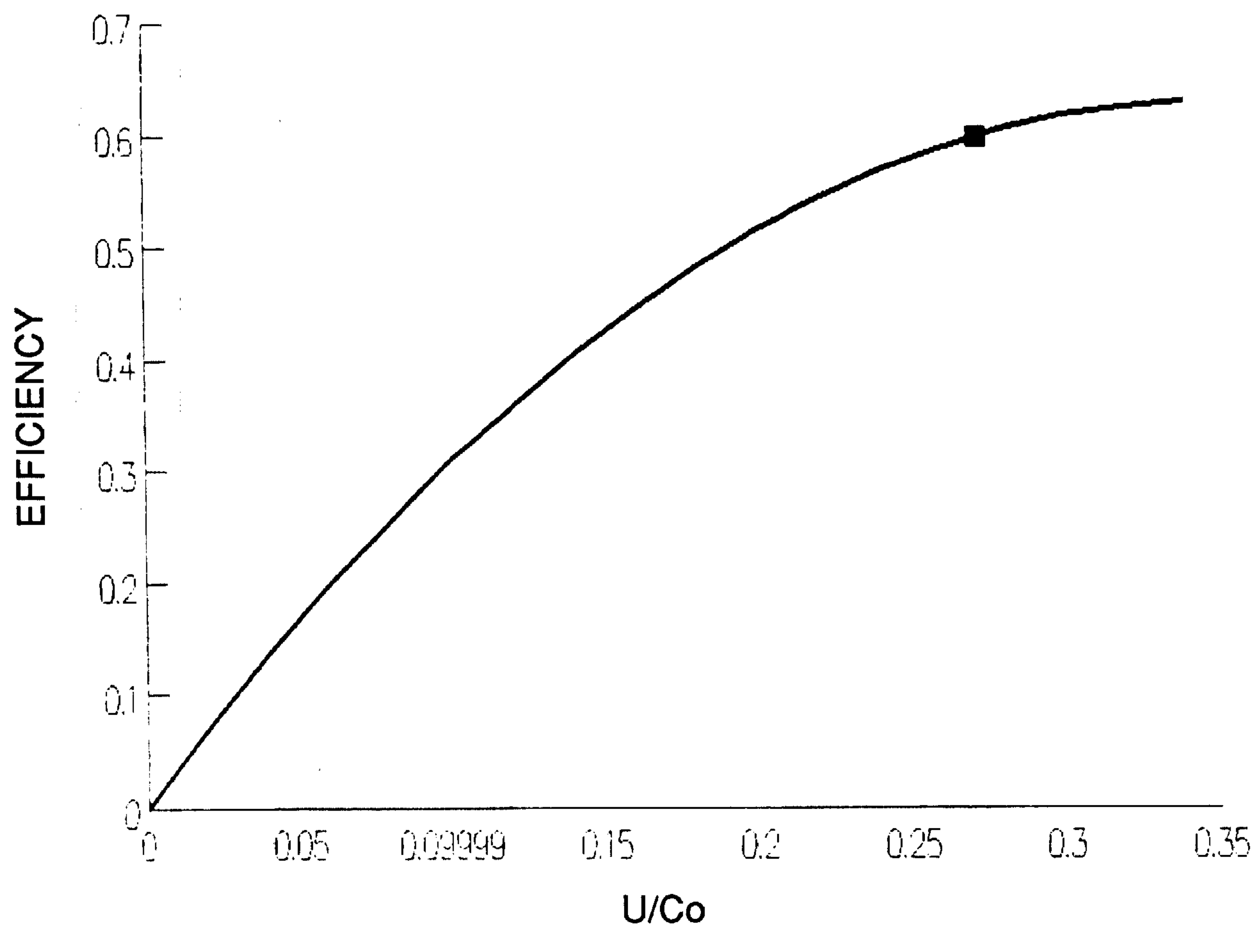
OUR HRB FULL ADMISSION TURBINE IS ALSO CHARACTERIZED



Following Selection of the Best HRB Propulsion Concept, We Performed Several Studies to Better Define the "Best" HRB Design Point. We Examined:

- HRB Shape (Tank and Motor Case Shapes) Effects on Drag and Weight
- Pressure Fed HRB Optimum Pc
- Turbopump Fed HRB Optimum Pc
- Turbopump Fed HRB Optimum MR—In This Simple Study, We Shifted the MR From the Peak I_{sp} Value of 1.8 to 1.9 (Liquid/Solid MR) and Gained Less Than 90.7 kg (200 lbm) of Payload to LEO. This Showed Us That We Had Very Nearly the Best MR Value, Which Also Improved Our Down-Throttling Performance Potential

DESIGN SENSITIVITY AND OPTIMIZATION STUDIES

Our Computer Studies Showed That the Important Mach Number Regions Are M = 1 to 4. This Is the HRB Operating Range Where Large Aerodynamic Pressures Arise During Flight. Our Biconic Design Drag Is Less Than a Conic HRB or the Smaller Aft Skirt Diameter Current SRB Over This Mach Number Range

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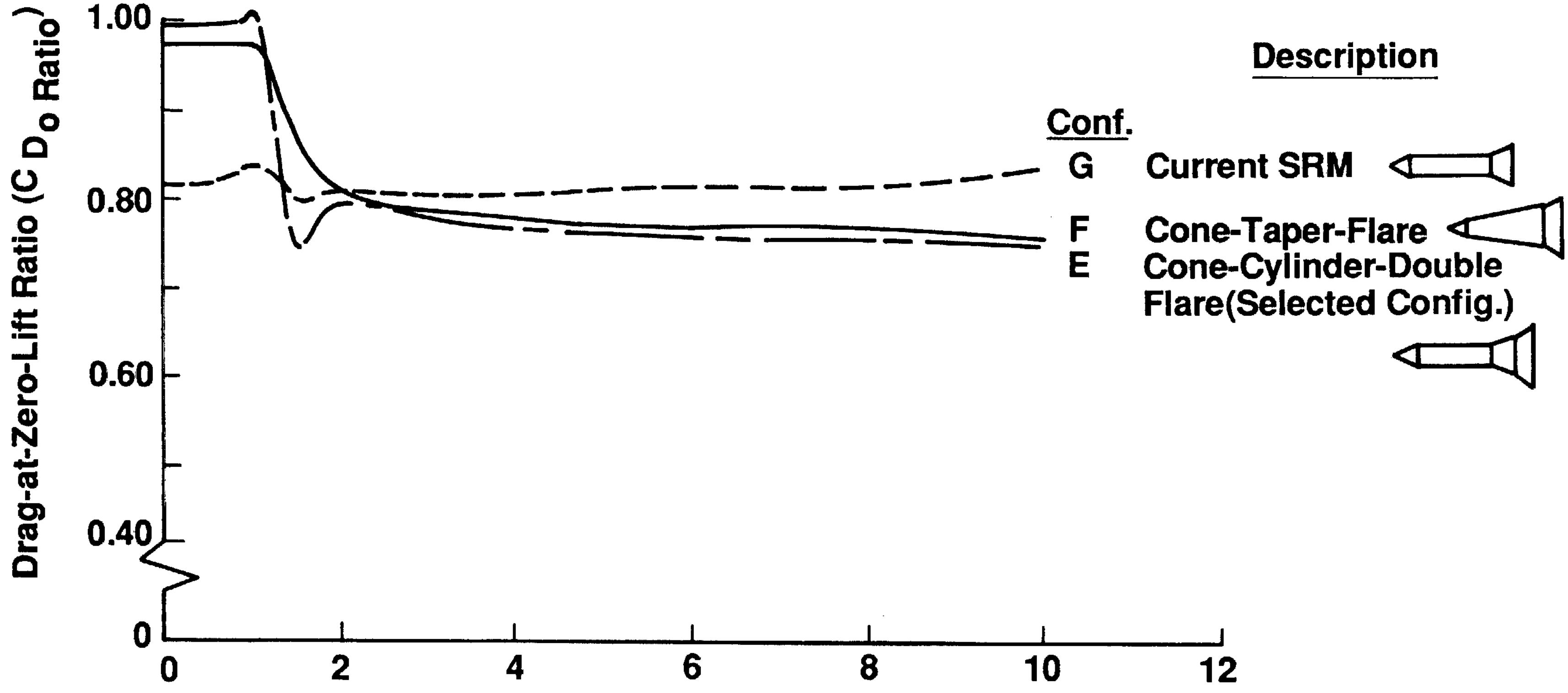
WE PERFORMED AN HRB AERO STUDY FOR SHAPE SENSITIVITY

C_{D_o} Ratio (1) vs Mach Number

•S_{REF} = 18.3 m^2 (197 ft^2) • Power-On • Missile DATCOM (Revised 11/85)



Notes: 1) C_{D_o} Ratio= C_{D_o} Conf./ C_{D_o} Baseline

2) Baseline Configuration is Cone-Cylinder-Flare

3) Drag=C_DQS

The Biconic Design Reduces Drag and Is Cost Effective. It Enables Mounting of Reasonably Sized Engines With Large Nozzles

WE SELECTED A CYLINDRICAL HRB WITH A TAPERED AFT END

Question	Answer
• Why Study Tapered HRB?	Trade Aerodynamic (Payload) Benefit of Tapering vs Production Cost Impact
• What Was the Result?	Tapering Entire HRB Is Not Cost-Effective
• Why?	Drag Is Dominated by the Aft Skirt, and Tapering Increases Costs
• What Should We Do?	Apply Aero-Drag Reduction Tapering to the Aft Skirt Region Only With Biconic Design
• Why Is This a Favorable Cost Trade?	Design Enables Use of Large Nozzles, Which Trades Favorably Via Reduced Propellant and HRB Weight, Size, and Cost. It Also Stiffens HRBs Against Launch and Prelaunch Bending Moments

We Selected the 15% Supplemental Fuel Film Cooling With Regeneratively Cooled Thrust Chamber:

1. It Gives the Best Specific Impulse Performance, Since It Has the Lowest Losses of Four Candidates. 15% Fuel Film Cooling (FFC) Yields the Lowest Nozzle/Drive Cycle Burnoff Loss, Because It Gives a Favorable MR in the Nozzle Boundary Layer. The Carbon-Carbon Throat Needs Only About 9% FFC to Prevent Erosion in 12.8 sec of Operation, but This Is Not Optimum. Silica Phenolic Needs 15% FFC to Prevent Gross Throat Growth. Thus, the Regeneratively Cooled Throat or the Carbon-Carbon Throat With 15% FFC Are Favored

2. It Is the Lowest Weight and Cost Solution, Because It Eliminates Two Turbine Drive Gas Generators Required to Prepare O_2-Rich Turbine Drive Fluid. It Also Eliminates Costly, Large Size Carbon-Carbon Throat Inserts. It Replaces Those Items With an Injector Face to Nozzle Area Ratio Three Tube Bundle, Which Functions as:

   - TCA Hard Wall Coolant Jacket

   - Turbine Drive Fluid Gas Generators

   - Autogenous LO_2 Tank Pressurization Heat Exchanger

THROAT COOLING OPTION SELECTION SUMMARY

FOM Options		Silica Phenolic Throat		Carbon-Carbon	(Highest Payload at Lowest Cost) Regenerative Tube Bundle 15%
Category	Criteria	9% FFC	15% FFC	Throat 9% FFC	FFC
Payload	Mission Avg. I_s Loss	~6%	2.5%	2.75%	2.2%
	Cause	Doubled Throat Area Burnoff Losses: • Low ε • Bad Contour • ~1/3 P_c 4.14 MPa (600 psia)	Nozzle Barrier Burnoff + Throat Growth Losses	Nozzle Barrier Burnoff Loss	MR Barrier Burnoff Loss
Payload	Hdw. Weight	High	High	High	Low
	Cause	2 Gas Generators	2 Gas Generators	2 Gas Generators	No Gas Generators
LLC	Cost	Medium	Medium	High	Medium
	Cause	2 Gas Generators	2 Gas Generators	Costly Throat Insert + 2 Gas Generators	Regen Tube Bundle Only

Our Pc Optimization Showed That Pc ≈ 9.65 MPa (1,400 psia) Gives the Same Payload at a GLOW Less Than 1% Less Than Our Baselined 6.9 MPa (1,000 psia) Pressure. This Validates Our Turbopump Fed HRB Selection

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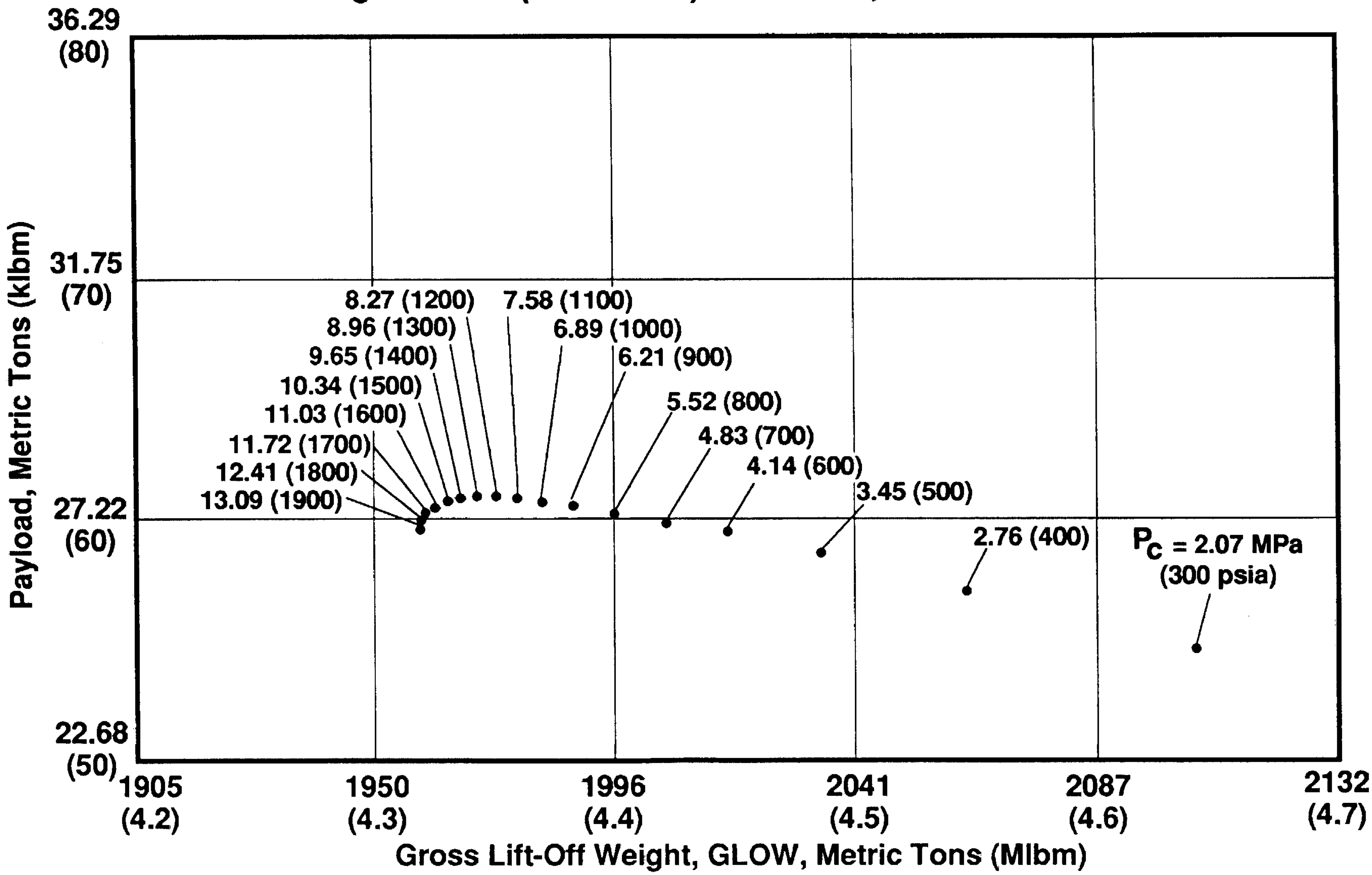
PRESSURE FED (PF) HRB, P_C EFFECT ON STS PAYLOAD AND GLOW

Propellant Grain #8. B Sat. HC (PEBC) + LO2
D_e = 3.8 m (149.6 in.) Baseline, 8 PF - 00



M13 HD-066

Our Turbopump Fed Combustor Pressure Optimization Study Showed a Peak Result of About 11.7 MPa (1,700 psia), Rather Than the 12.4 MPa (1,800 psia) We Had Baselined. Thus, Our Design Point Is at Pc = 11.7 MPa (1,700 psia)

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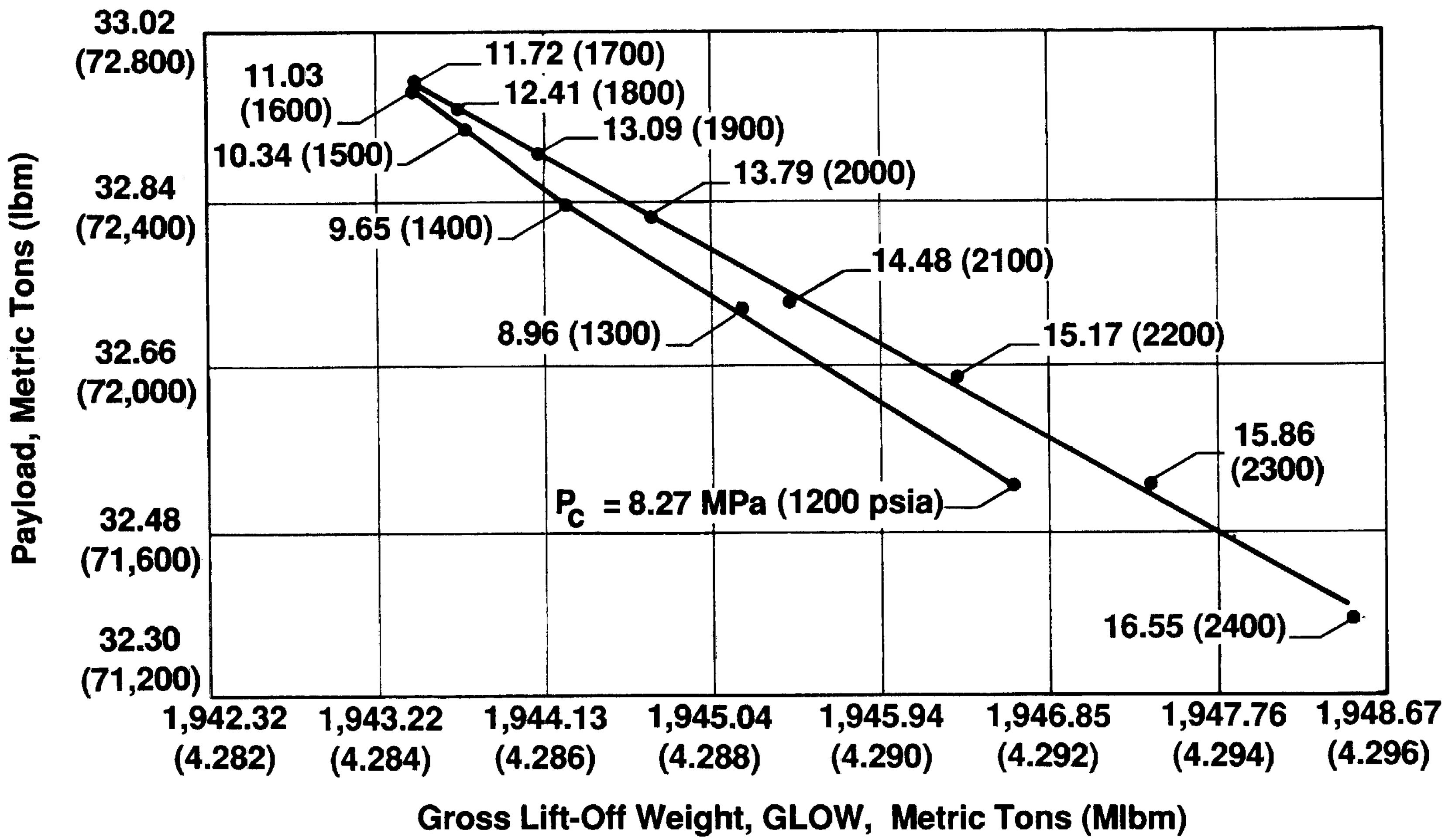
TURBOPUMP FED HRB OPTIMIZATION OF P_C FOR 8 D EBBC - 2



M7/D13/H-1

This Plot Expands the Sensitivity of the P_c vs GLOW Data We Generated. It Shows That a Minimum Weight System Is Achieved With 11.7 MPa (1,700 psia) Combustor Pressure

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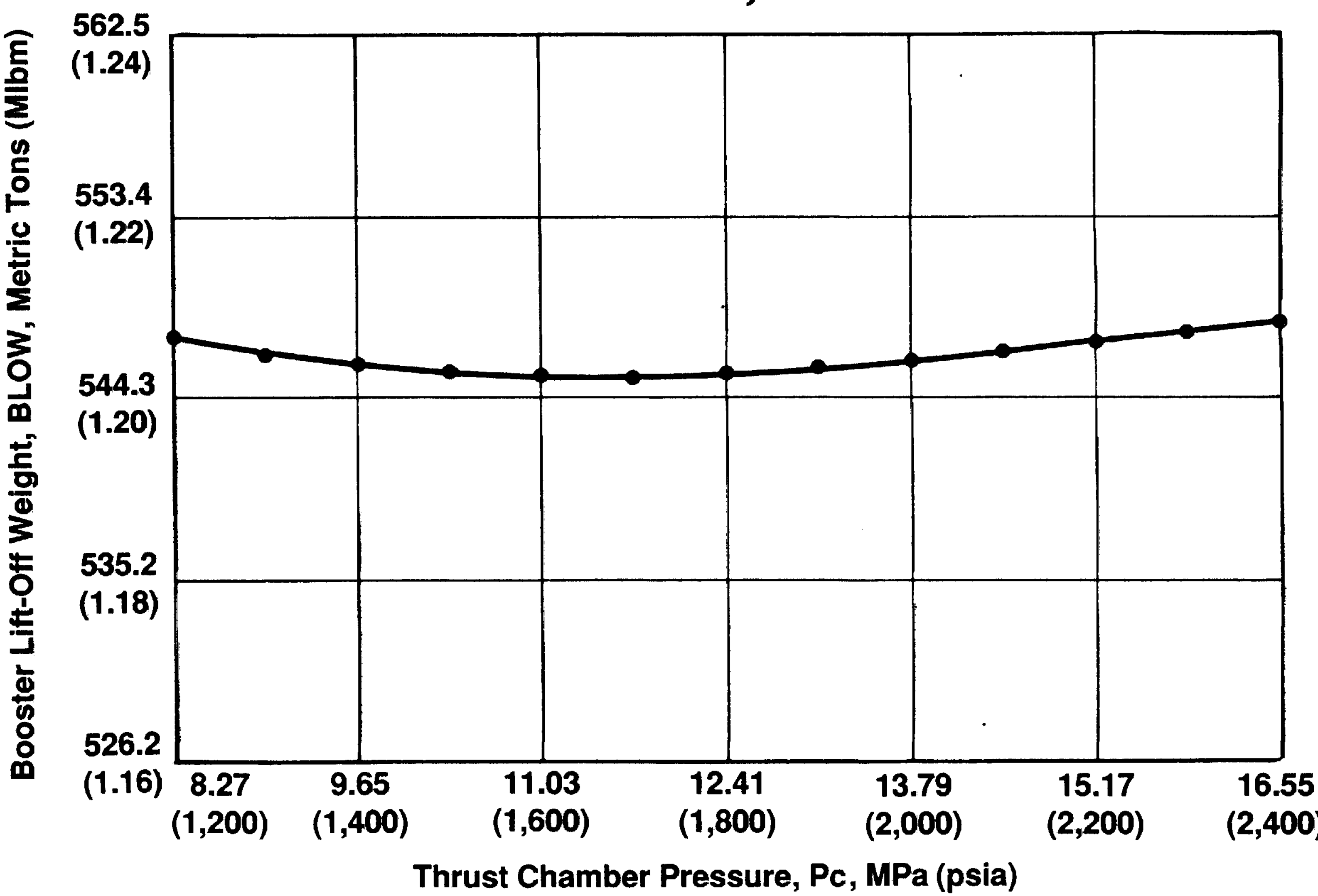
TURBOPUMP FED HRB OPTIMIZATION OF PC FOR 8 D EBBC - 2 RESULTING BOOSTER LIFT-OFF WEIGHT, BLOW, VS PC



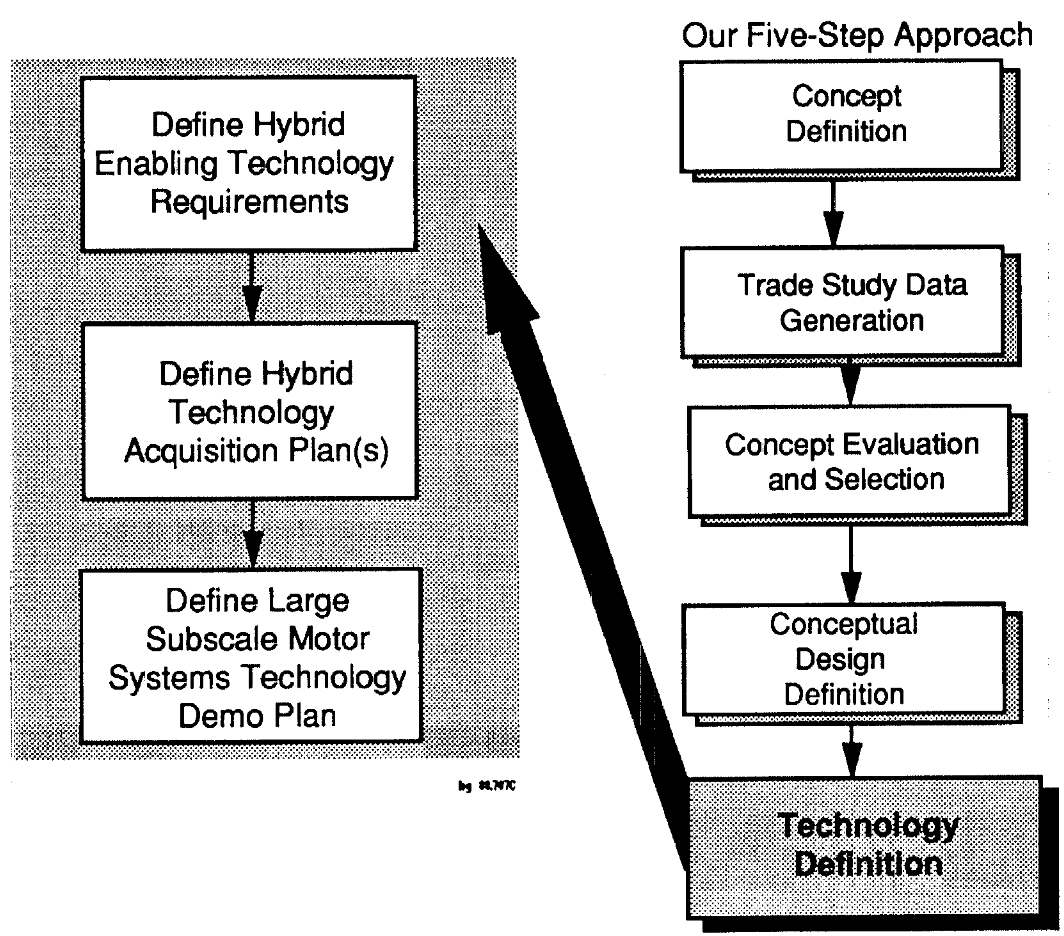
C5

In Task 5, Technology Definition, We:

- Define HRB Design Concept-Enabling Technology Requirements
- Define Our HRB Technology Acquisition Plans
- Define Our Large Subscale Propulsion System Demonstration Plan

Our Results Will Be Presented in This Order

Technology Definition



This Table Summarizes Our Phase II HRB Enabling Technologies and Justifies Each on the Basis of Payload and Cost Benefits. The Reasons Why the Technologies Are Considered to Be Low Risk Are in Parenthesis With the Technology Statements

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OUR HRB TECHNOLOGY DEFINITION LIST IS SHORT, LOW RISK, AND WELL JUSTIFIED

Technology	Benefit
1. Solid Propellant Gas/Liquid Injector (Gas/Liquid Injectors Successfully Tested)	>15% Higher Combustion Efficiency vs Forward Injection, Improves I_{sp}, Weight, Cost, and Payload
2. Fuel Rich Propellant and Ignition (Similar Propellant Successfully Tested)	Provides for Reduced LCC and ~20% P/L Advantage of SLSC Concept
3. Fuel Rich Gas Control Plates (Routine With Fixed Plates)	Improves Both P/L and Cost • Allows Safe Abort With TCA Out • Provides Independent MR Control for Improved Propellant Utilization • Increases Isp by Providing Uniform Gas Flow to Injectors • Aids Ignition
4. GO_2 Bleed Burnoff in Nozzle (Routine Without Combustion)	Improves Both P/L and Cost • Renders Low Cost Cycle Feasible • Reduces Turbine Bleed I_{sp} Loss • Protects Flex Seal

TECHNOLOGY REQUIREMENTS

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WE HAVE SELECTED AND PRIORITIZED OUR HRB TECHNOLOGY

Priority	Technology	Benefit
1	Solid Propellant Gas/Liquid Injector (Gas/Liquid Injectors Successfully Tested)	>15% Higher Combustion Efficiency vs Forward Injection; Improves I_{sp}, Weight, Cost, and Payload
1	Fuel-Rich Propellant and Ignition (Similar Propellant Successfully Tested)	Provides for Reduced LCC and ~20% P/L Advantage of SLSC Concept
2	Fuel Rich Gas Control Plates (Routine With Fixed Plates)	Improves Both P/L and Cost: • Allows Safe Aborts With TCA Out • Provides Independent MR Control for Improved Propellant Utilization • Increases I_{sp} by Providing Uniform Gas Flow to Injectors • Protects Injector • Reduces Development Cost (Ignition and Stability)
3	GO_2 Bleed Burnoff in Nozzle (Routine Without Combustion)	Improves Both P/L and Cost • Renders Low Cost Cycle Feasible • Reduces Turbine Bleed I_{sp} Loss • Protects Flex Seal and Cools Nozzle

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SOLID PROPELLANT GAS/ O_2 INJECTOR TECHNOLOGY

- Our SLSC Concept Score (LCC + P/L) Was 90% Higher Than the Next Best Hybrid Concept Without an Aft Injector (36 Points vs 19) for the Low Launch Rate Scenario and More Than 75% Higher for High Launch Rates (46 vs 26)

- Our SLSC Concept Requires an Aft Injector to Add O_2 to the Fuel-Rich Solid Propellant Gas to Provide Good Combustion Efficiency and High Performance

- Gas/Liquid Injector Combustion With Gas Generated by a Solid Propellant Will Be Demonstrated

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FUEL-RICH SOLID PROPELLANT TECHNOLOGY

- Aft Injector Concept Scores (LCC + P/L) Are 66 to 99% Higher Than Those With Only Forward Injectors

- All HRB Propulsion Concepts That Meet NASA Requirements and Have Aft O_2 Injectors Require Fuel-Rich Solid Propellant, Rather Than Pure Fuel Solid Grains

- We Will Demonstrate Fuel-Rich Solid Propellant Ignition and Combustion That Will Meet All NASA HRB Requirements

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FUEL RICH SOLID PROPELLANT GAS CONTROL PLATE TECHNOLOGY

I.
- Scores (LCC + P/L) for Large HRBs (2/Launch) Are 70 to 85% Greater Than for Small HRBs (8/Launch)
- Multiple (4) TCA Design Scores Were 28 to 29% Greater Than Single TCA Designs
- Variable Flow Area Control Plates Will Assure Safe Abort Capability to Large HRBs With Multiple TCAs

II.
- Variable Flow Area Control Plates Provide O/F Mixture Ratio and Propellant Control Independently of Thrust Modulation for Each TCA Separately Which Increases HRB Payload Performance
- Variable Flow Area Control Plates With Solid Propellant-Generated Fuel-Rich Gas Will Be Demonstrated

GO_2 BLEED BURNOFF TECHNOLOGY

- Turbopump-Fed HRBs Scored Higher Than Pressure-Fed HRBs by 39 to 43% With Bleed Cycle Turbine Drives
- The Oxidizer-Rich Turbine Drive Concept Is Only Feasible if Turbine Exhaust Can Be Burned With Fuel-Rich Thrust Chamber Barrier Coolant in the Rocket Nozzle
- Our Oxidizer-Rich Turbine Drive HRB Score (LCC + P/L) Was High Because the I_s Was Increased by Burning the Turbine Gases With the Fuel-Rich TCA Coolant Boundary Layer, and an Expensive C-C Nozzle Insert Was Unnecessary
- The Expander Bleed Burnoff Turbine Drive Cycle Was Selected Using Supercritical LO_2 Regeneratively Cooled Thrust Chamber From the Injector to Nozzle Area Ratio of 3.0

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TECHNOLOGY ACQUISITION

• Plan • Facilities • Schedule • Cost	for	• Propellant and Igniter • Injector • Control Plate • GO_2 Bleed Burnoff

PROPELLANTS

Existing Formulations Are Available, and Have Been Tested at Aerojet, That Provide Adequate Fuel Rich Grain Characteristics for the SLSC Hybrid. The Initial Phase II Testing 31 kN (7 klbf Thrust), BATES Motor Will Use These Demonstrated Formulations.

Optimization of a New Formulation Is a Recommended Technology for Phase II for Increased Performance, Reduced Combustion Temperature (Increases Reliability and Reduces Injector Heating), and Carbon Reduction.

Further, While Not a Contractual Requirement, Aerojet Has Defined Reduction of HCl as a Technology Requirement for Future Solid Propellants. The Formulation That We Will Select for Phase II Evaluation Will Have Less Than 1% HCl (Preferably 0%) in Its Exhaust

WE HAVE IDENTIFIED THE PROPELLANT REQUIRED FOR SLSC WITH CLEAN EXHAUST

Our Selected Fuel Rich Solid Propellant Contains a Scavenger Oxidizer System for Low HCl Exhaust Products and the High Hydrogen Content TPE Polymer Binder. The Composition of This Phase I Selected Propellant Is Shown Below.

Ingredient	Weight %
Ammonium Perchlorate Oxidizer (AP)	20.31
Sodium Nitrate Scavenger (SN)	14.69
Polyethylene Block Copolymer (PEBC)	65.00
	100.00

Performance of Our Solid Propellant Depends on Its Capability to Deliver an Adequate Supply of Fuel Rich Gas to the Aft Combustion Chamber for Efficient Oxidation With LOX. One Key to Maximum System Performance Is to Develop a Fuel-Rich Solid Grain With the Minimum Solid Oxidizer Necessary to Sustain Combustion in the Primary Chamber.

A Scavenger Oxidizer That Has Been Developed and Demonstrated Has Been Selected as the Baseline Oxidizer for the Hybrid Fuel Rich Grain. This Oxidizer System Provides for Utilization of the Well Characterized Ammonium Perchlorate (AP) With Sodium Nitrate (SN) Added to Neutralize the HCl Exhaust to Salt.

$$NH_4ClO_4 + NaNO_3 \longrightarrow NaCl + \text{Nontoxic Products}$$

Phase I Theoretical Studies Have Shown That System Performance Is Maximized When the Oxidizer Is Minimized in the Fuel Rich Grain. Thus a Key Technology Is to Determine Experimentally the Minimum Amount of $NH_4ClO_4/NaNO_3$ Oxidizer Required to Support Combustion in the Primary Chamber

SOLID PROPELLANT—TECHNOLOGIES

Key Technologies Associated With Development and Demonstration of the SLSC Solid Propellant Grains Are

- Performance Properties
  I_{sp}, Combustion Efficiency, Combustion Temperature
- Processing Properties
  Viscosity, Pot Life, Time to Zero Flow
- Ballistic Properties
  Burning Rate, Pressure Exponent, Stability, Ignition, Extinguishment (Abort)
- Mechanical Properties
  Response Properties, Failure Properties
- Hazard Properties
  Material, Processing, Military/DOT Classification
- Aging Properties
  Accelerated, Long Term
- Environmental Properties
  Exhaust Composition

SOLID PROPELLANT ACQUISITION PLANNING

Technology	Variable	Testing	Data
Performance	• Oxidizer Type, Size, and Ratio • Polymer and Secondary Fuel Type and Particle Size	• Liquid/Solid Strand and 0.23 kg (0.5 lb) BRM* • Liquid/Solid Strand and 0.23 kg (0.5 lb) BRM, Combustion Residue, Cone Burner	• Ignition, Burning Rate, Pressure Exponent, Extinguishment • Ballistics, Combustion Efficiency, Extinguish-ment
Processing	• Solids, Viscosity, Pot Life, Time to Zero Flow	• Roto Viscometer • Zero Flow Tubes	• Viscosity, Pot Life, Time to Zero Flow
Ballistics	• Total Solids, Particle Size Distribution, Combustion Catalysis Effects on Burn Rate and Pressure Exponent • Temperature Sensitivity Tests on Selected Composition • Effect of Total Solids, Particle Size, Stability Catalysts, and Acoustic Cavities on Combustion Stability	• Liquid/Solid Strands, 0.23 kg (0.5 lb) BRM • π_k AFAL Test Motors (at AFAL) • T-Burner Pulse Motor 0.45 kg (1.0 lb) • Center Vent Pulse Motor 0.45 kg (1.0 lb)	• Burn Rate vs Pressure • Burn Rate at Various Pressures • Effect of Catalysts • Effect of Temperature on Chamber Pressure • Combustion Stability Sensitivity to Pressure and Velocity Coupled Instability
Mechanical Properties	• Total Solids and Particle Size, Wetting/Bonding Agents, Curing Agent, Mix Cycle, Cure Cycle	• Response Properties, Stress Relaxation, and Coefficient of Thermal Expansion, 25 gm and 100 gm Samples (0.055 lbm and 0.22 lbm)	• Failure Properties—Strength, Elasticity • Safety Margins

*BRM = Burning Rate Motor

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SOLID PROPELLANT ACQUISITION PLANNING
(CONTINUED)

Technology	Variable	Testing	Data
Hazard Properties	• Oxidizer Type and Particle Size/ Distribution, Combustion Catalysts, Secondary Fuel Type	• Impact Friction, Heat, Spark, Detonability	• Hazard Assessment and Classification Rating (1.4)
Environmental	• Scavenger Type With Varying Levels of KNO_3, KN_3, $NaNO_3$, NaN_3, $LiNO_3$, LiN_3	• Closed Bomb Exhaust Collection and Analysis for HCl and BRM IR Plume Tests	• Exhaust Composition (HCl)

IGNITER

Classic Hybrid Designs Have Introduced the Oxidizer Into the Grain Providing an atmosphere That Was Conducive to Ignition. The SLSC Hybrid, However, Uses a Fuel-Rich Grain With the Oxidizer Injection Only in the Secondary Combustor

The Fuel-Rich Grain Requires an Ignition Source Where:

- The Temperature of the Surface (or a Thin Adjacent Layer) Must Be Raised to an "Ignition Temperature" That Is Characteristic of the Propellant
- A Thermal Gradient Must be Established at the Propellant Surface
- The Environmental (Chamber) Pressure Must Exceed the Low Pressure Ignition Limit
- The Temperature of the Igniter Combustion Product Must Be Less Than Some Critical Value (to Be Determined) That Could Cause Thermal Failure of the Injector and Associated Hardware
- The Combustion Product Must Not Contain Chemical Species That May Condense and Be Deposited on the Injector and Associated Hardware
- The Igniter Mass Flow Rate vs Time Function Must Be Specified (Based on Ignition Transient Gas Dynamic Analysis) to Minimize the Possibility of Significant Pressure Waves

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FOUR CANDIDATE IGNITERS ARE AVAILABLE FOR THE SLSC HYBRID

Type	Charge	Features
Pyrotechnic (Propellant Charge)	Initiator With AlClO Pellets and Propellant Grain Charge Formulated to Be Oxidizer Rich	Requires Ignition Source and S/A Device. Simple System With No Propellant Tanks Required (See Below). Less Available Oxidizer to Assist in Fuel-Rich Grain Ignition
Hypergolic	Hypergolic Liquid Propellants Such as N_2O_4/MMH Operating in an Oxidizer Rich Regime	No Ignition Source Required. Temperature Controlled by Mixture Ratio. Oxidizer-Rich Hot Gas Stimulates Ignition of Fuel-Rich Grain. No Particles in Exhaust. Require 2 Tanks, Valves, Propellant Lines, and Controls
Pyrophoric	Triethyl Aluminum (TEA) With LO_2 Operating Oxidizer Rich	Same as Hypergolic With System Reduction to One Tank. LO_2 Available From Booster Tank
Hot Gas	Oxygen/Gaseous Hydrocarbon	Similar to Hypergolic But Requires an Ignition Source. Good Dispersal in Solid Grain

We Have Performed Preliminary Analyses on Two Types of Hybrid SLSC Ignition Systems. The Hot Gas and Pyrotechnic Igniters Will Both Ignite the Fuel-Rich Grain but Need Further Analysis and Bench Tests for Evaluation.

The Pyrophoric and Hypergolic Igniters Have Yet to Have a Preliminary Analysis but Their Actions Have Been Characterized in Other Systems, and Elevation to the Same Status as the Pyrotechnic and Hot Gas Will Be Easily Performed at the Start of Phase II

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HOT GAS IGNITER

A Convenient Gaseous Energy Source May Be Obtained as the Result of Combustion of Oxygen and a Gaseous Hydrocarbon Fuel. Gaseous Nitrogen May Be Introduced to the Mixture as a Diluent to Reduce the Temperature of the Combustion Product. The Combustible Mixture May Be Over-Oxidized to Provide an Oxygen Rich Combustion Product to Enhance Ignition of the Motor Propellant Surface. As an Example, Fuel Was Selected as Gaseous Ethyne [Acethylene, C_2H_2 (g)] Which Was Combusted With O_2 (g), Using N_2 (g) as a Diluent. The Resulting Flame Temperatures for Various O/F Ratios (by Weight) Are Presented in the Following Table.

As With the Propellant Charge Igniter, Flame Temperature and Flow Rate Would be Selected on the Basis of Satisfying the Motor Propellant Ignition Requirements, Ignition Ballistic Specifications, and Results of Ignition Gas Dynamic Analysis. The O/F Ratio and Diluent Mass Fraction Would Be Selected So That Flame Temperature Is Less Than Critical for Thermal Integrity of the Injector

O/F -->	3.333 (O/F Near Balanced)				5.000 (O_2 Rich)				6.667 (O_2 Rich)			
O_2, %	35.00	28.00	21.00	14.00	40.00	32.00	24.00	16.00	42.50	34.00	25.50	17.00
C_2H_2, %	15.00	12.00	9.00	6.00	10.00	8.00	6.00	4.00	7.50	6.00	4.50	3.00
N_2, %	50	60	70	80	50	60	70	80	50	60	70	80
T_f, °K (°R)	3406 (6130)	3007 (5413)	2403 (4326)	1683 (3029)	3149 (5668)	2777 (4999)	2231 (4017)	1556 (2801)	2667 (4801)	2238 (4029)	1740 (3133)	1188 (2139)

PYROTECHNIC IGNITER

This Type of Igniter Utilizes a Solid Propellant Grain as the Ignition Heat Source. The Solid Propellant Charge Is Ignited and Burned in a Vented Pressure Vessel. The Propellant Grain and Chamber Are Designed to Provide the Mass and Energy Flow Rates to Ignite the Motor Chamber Propellant Surface, Based on the Ignitability Test Results, the Ignition Ballistics Specifications, and the Results of Ignition Gas Dynamics Analysis. The Igniter Propellant Charge Will Be Formulated (O/F Ratio) So That the Flame Temperature Is Less Than the Critical Value for Thermal Integrity of the Injector. This Critical Value Will Be a Function of the Mass Flow Rate Through the Injector Passages, as Well as Injector Material Properties and Thermal Design. As Ignition of the Solid Propellant Surface Is Also a Function of Both Igniter Gas Flame Temperature and Mass Flow Rate, the Ignition System Analysis Must Interact With the Injector Thermal Analysis to Determine Optimum Igniter Mass Flow Rate and Flame Temperature.

Igniter Combustion Product Gas Temperature vs Fuel/AP Ratio

Fuel, %	10	15	20	25	30
AP, %	90	85	80	75	70
T_f, °K (°R)	3083 (5549)	2692 (4846)	2139 (3850)	1611 (2899)	1109 (1997)

PLANNING

Igniter Development Will be Initiated at the Completion of Propellant Laboratory Testing and Will Be Designed to Fulfill the Ignition Criteria of the Final Propellant Formulation.

The Solid Propellant Igniter Performance Will Be Verified by Means of Laboratory Bench Tests. Subscale Igniters Would Be Discharged Into Test Chambers Scaled to Simulate the Actual Motor Chamber With Respect to Heat Flux and Pressurization. Instrumentation Will Include Calorimeters to Measure the Heat Flux as a Function of Position in the Chamber, Thermocouples to Record the Gas Temperature, and Pressure Transducers to Record Pressure at Several Test Chamber Locations.

The Subscale Test Chamber Will Contain Samples of the Motor Chamber Propellant to Determine the Extent That an Oxygen Rich Igniter Gas Enhances Ignitability

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PHASE II TECHNOLOGY ACQUISITION PLAN

- Introduction and Summary
- Plans/Schedules
- Costs

ACQUISITION PLANS/SCHEDULES

Our Plan Is to Design, Fabricate, Inspect, and Test These Technology Hardware Items in Two Test Series. The First Test Series Will Use Available, Standard BATES (Ballistic Test Evaluation Standard) Motor Hardware, and Will Build a Small Vaned Injector. Testing Will Be Done With Currently Available Fuel-Rich Propellant and Will Demonstrate SLSC Combustion Operation and Performance at the 31 kN (7 klbf) Thrust Level for Short Durations (4 seconds).

The Second Test Series Will Use the Significantly Larger, Standard Super BATES Motor, a Larger, Involute-Vaned Injector With Matching Control Plate and Nozzle GO_2 Injection Hardware. Testing at 350 kN (80 klbf) Thrust Will Be Performed for Longer Durations (12 Seconds) to Confirm Scale-up Operation and Performance, Control, and Longevity. The Table Below Shows How Our Plan Minimizes Thrust Scale-up Risk for Larger Hardware, While it Increases Test Durations.

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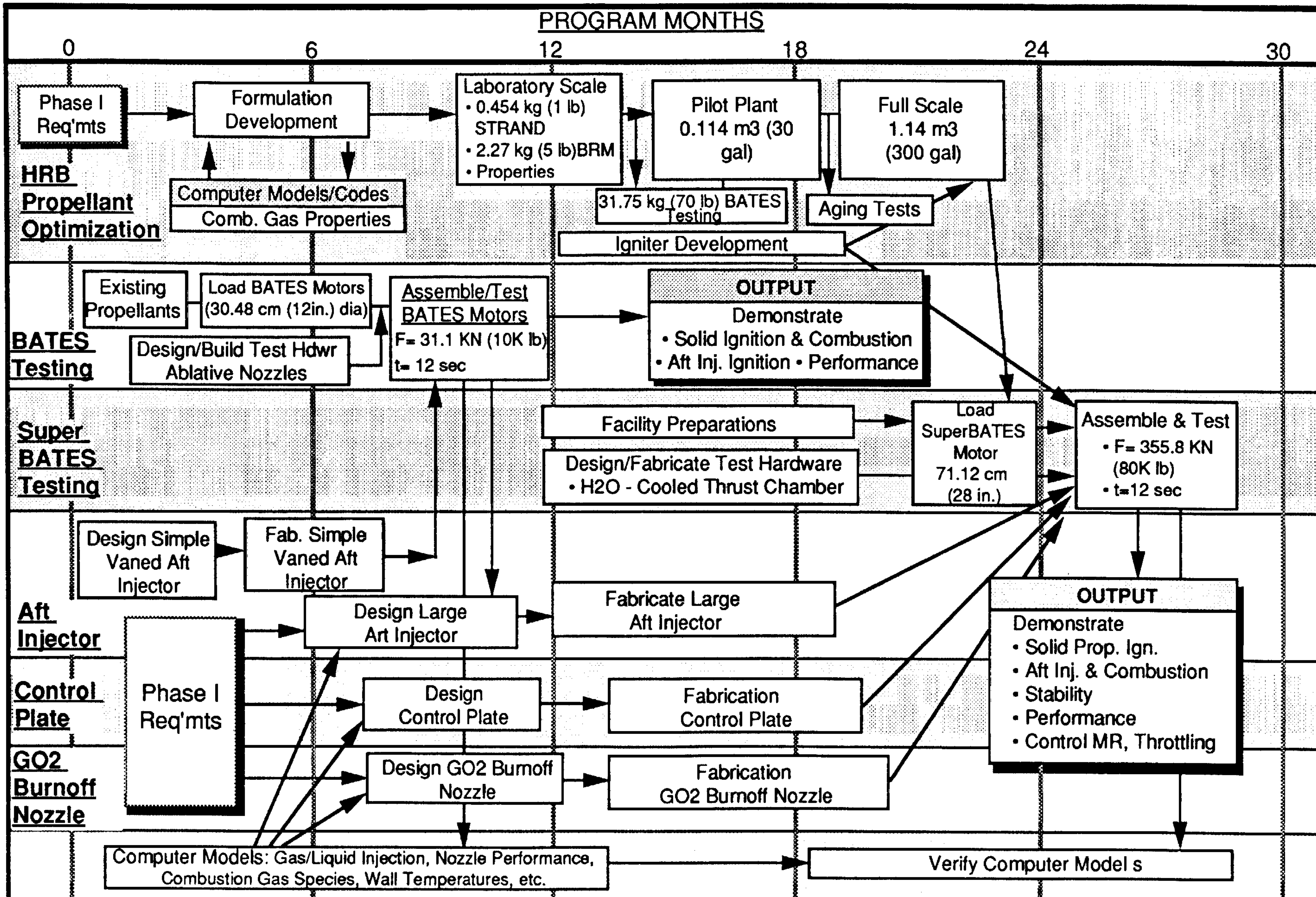
OVERALL SUMMARY—A LOGICAL SCALE-UP, LOW RISK APPROACH TO HRB TECHNOLOGY DEMONSTRATION

HRB Project Phase	Engine Vacuum Thrust Level, (klbf)	Thrust Scale-Up Ratio	Test Duration, sec	Duration Scale-Up Ratio	Solid Case	Purpose
II . a	31 kN (7)	-----	4		BATES Motor 0.305 m (12 in. Dia)	• Solid Propellant • Injector (Performance)
		8.0		3.0		
II . b	350 kN (80)	-----	12		Super BATES 0.711 m (28 in. Dia)	• Solid Propellant • Gas Control Plate • Bleed Burnoff (Performance)
		5.0		3.0		
III	1.8 MN (400)	-----	36		Stage II Peacekeeper 2.34 m (92 in. Dia)	• Cold GO_2 Turbine • LO_2 Cooled TCA • SS Splitline TVC • Hoop Wrapped Coni-Cyl Case
		2.0		3.5		
Development and Production (Large HRB for STS)	3.6 MN (800)	-----	128		Production 3.71 m (146 in. Dia)	• Flight

Risk Is Reduced With Decreasing Scale-Up Ratios

PHASE II ACQUISITION PLAN/SCHEDULE



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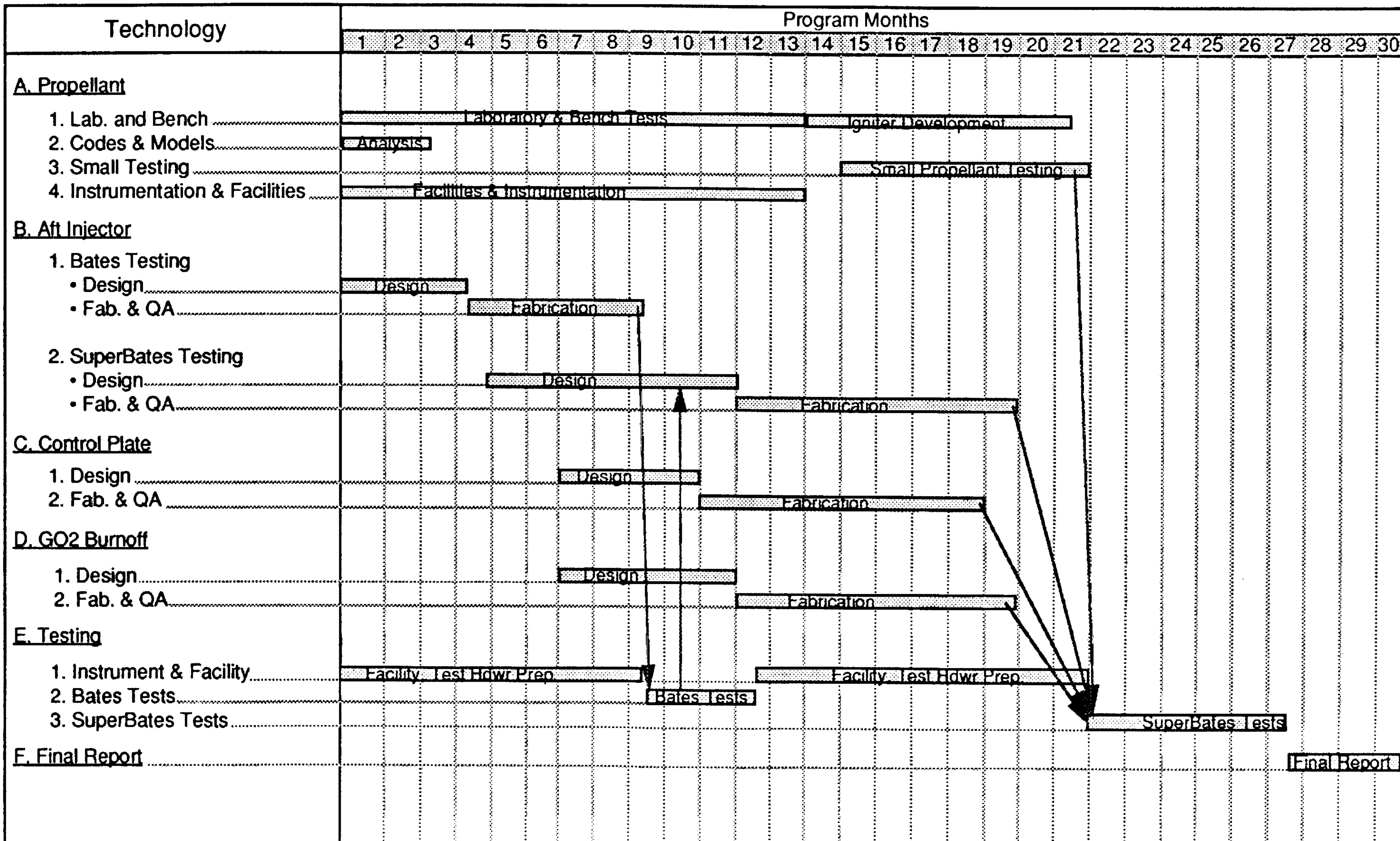
PHASE II - ACQUISITION SCHEDULE

OVERALL PROGRAM



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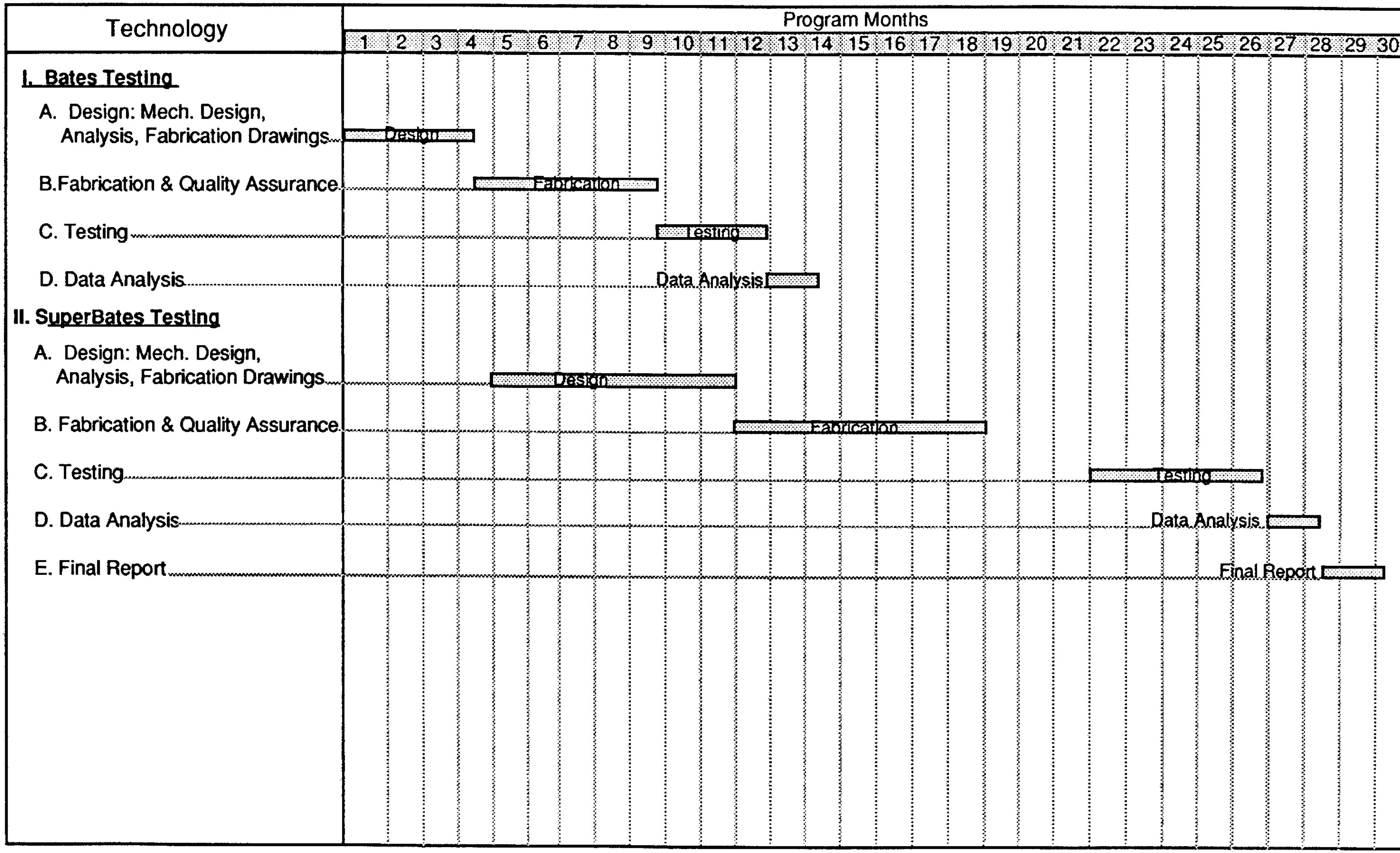
PHASE II - ACQUISITION SCHEDULE

AFT INJECTOR



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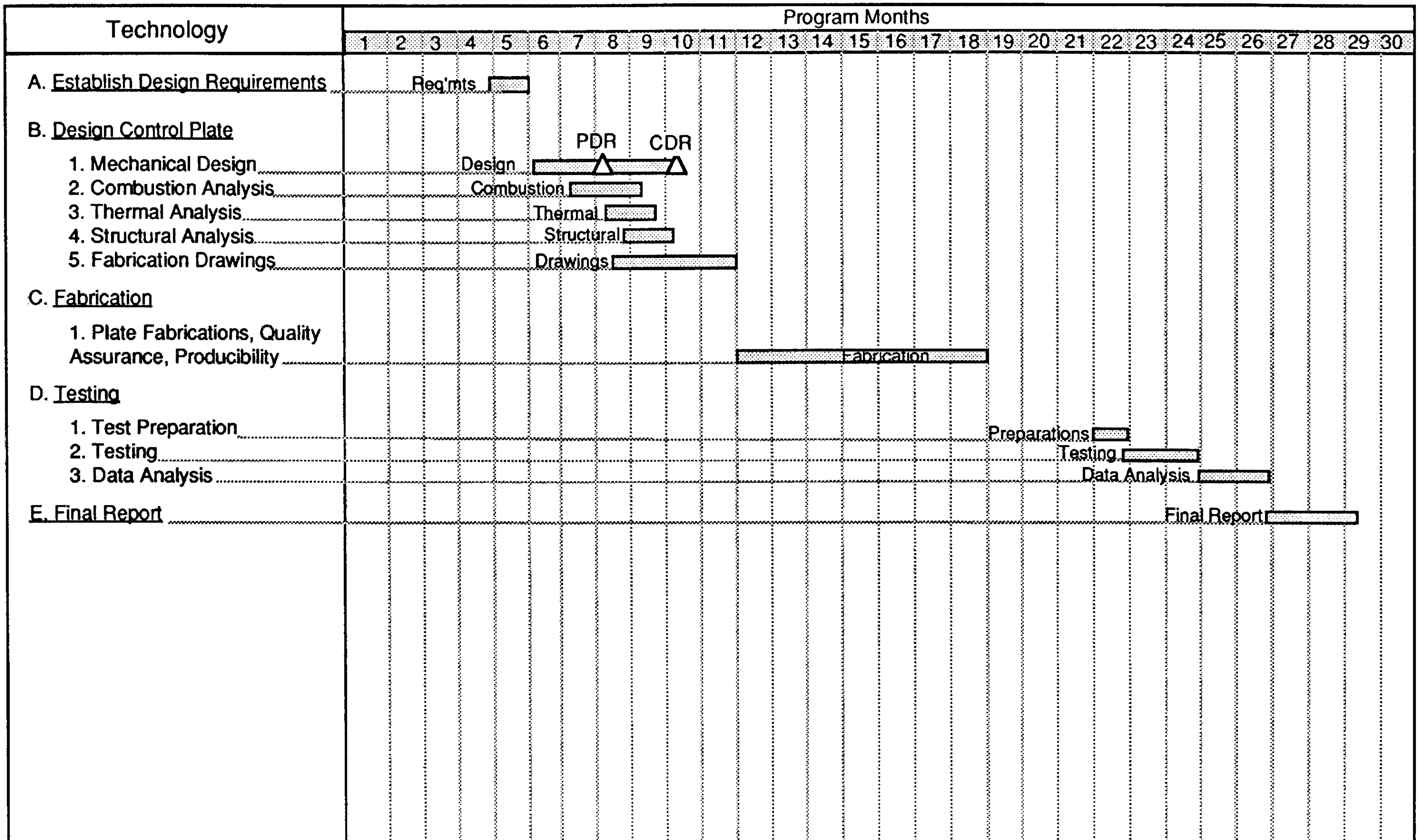
PHASE II - ACQUISITION SCHEDULE

CONTROL PLATE



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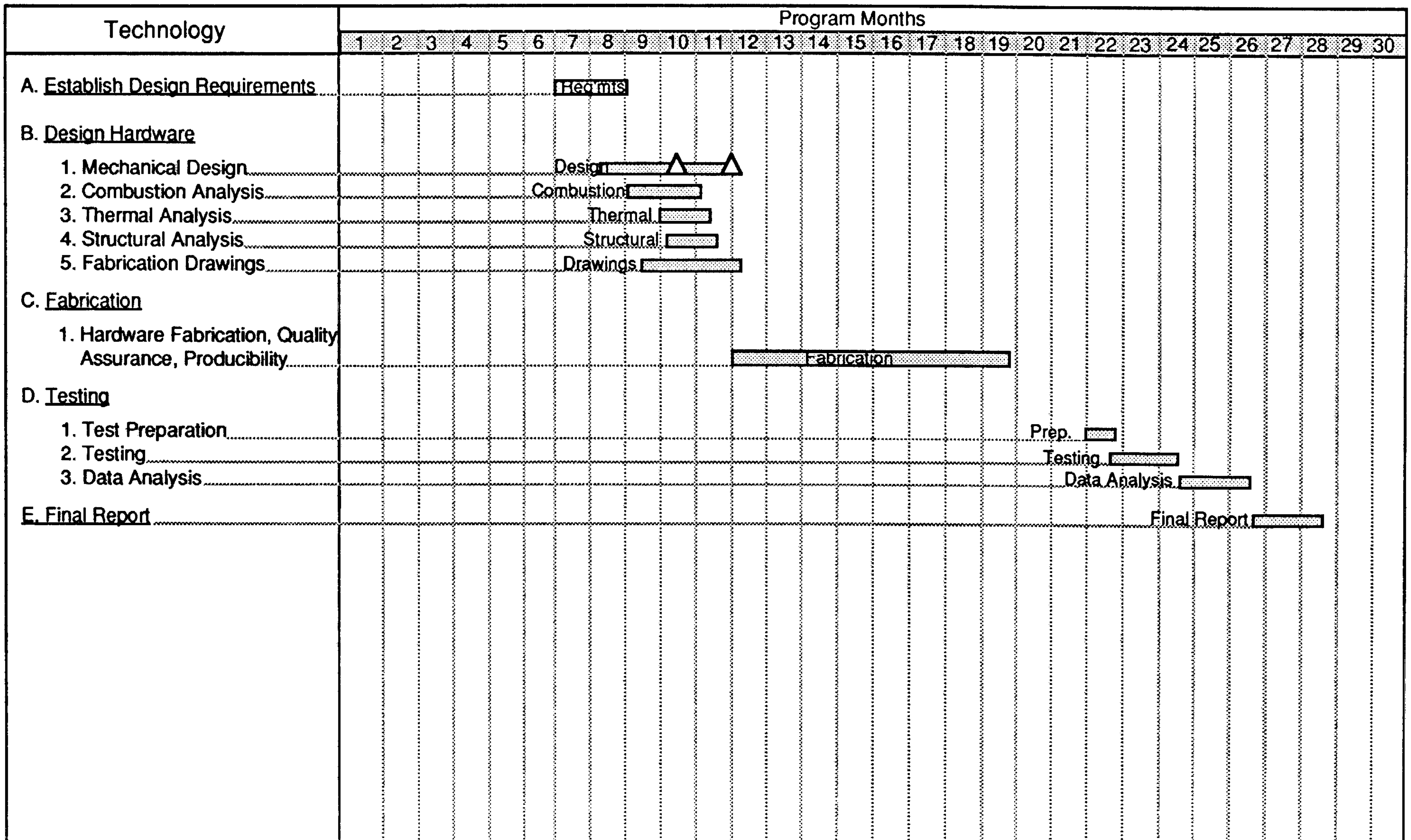
PHASE II - ACQUISITION SCHEDULE

GO2 BURNOFF



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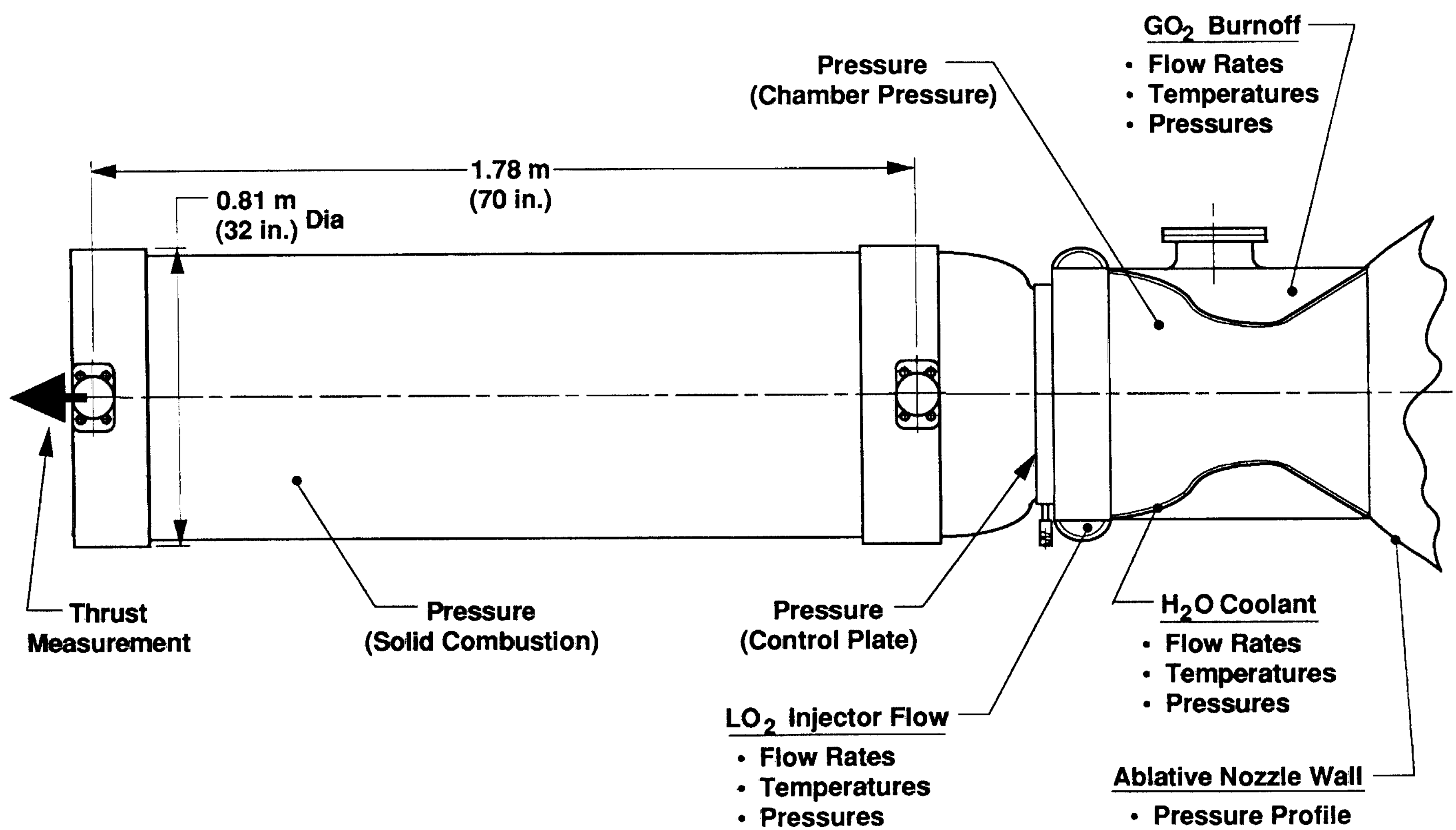
PHASE II - ACQUISITION SCHEDULE

BATES and SuperBATES Testing Typical Instrumentation Required



ACQUISITION COSTS

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PHASE II TECHNOLOGY ACQUISITION

Program Cost Summary

Technology	ROM Cost
• Propellant Optimization	$1.14M
• Igniter	$0.60M
• Aft Injector	$1.87M
• Control Plate	$1.00M
• GO_2 Burnoff Nozzle	$1.13M
• Testing (BATES and SuperBATES)	$2.26M
Total	$8.00M

PHASE II TECHNOLOGY ACQUISITION

ROM Costs

	HRB Enabling Technology		
Tasks	Aft Injector	Control Plate	GO_2 Burnoff Nozzle
Design	$0.544M	$0.318M	$0.385M
Fabrication	$1.546M	$0.682M	$0.745M
Total	$2.09M	$1.00M	$1.13M

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PHASE II TECHNOLOGY ACQUISITION

Testing Costs

Task	ROM Cost
1. BATES Motor Testing	
• Test Hardware Preparation (Load Motors, Fabricate Nozzles, Assemble Hardware)	$0.12M
• Facility Preparation/Testing	$0.17M
2. SuperBATES	
• Test Hardware Preparation (Design/Fabrication H_2O-Cooled Chamber, Facility Preparation)	$1.21M
• Facility Preparation/Testing	<u>$0.76M</u>
Total	$2.26M

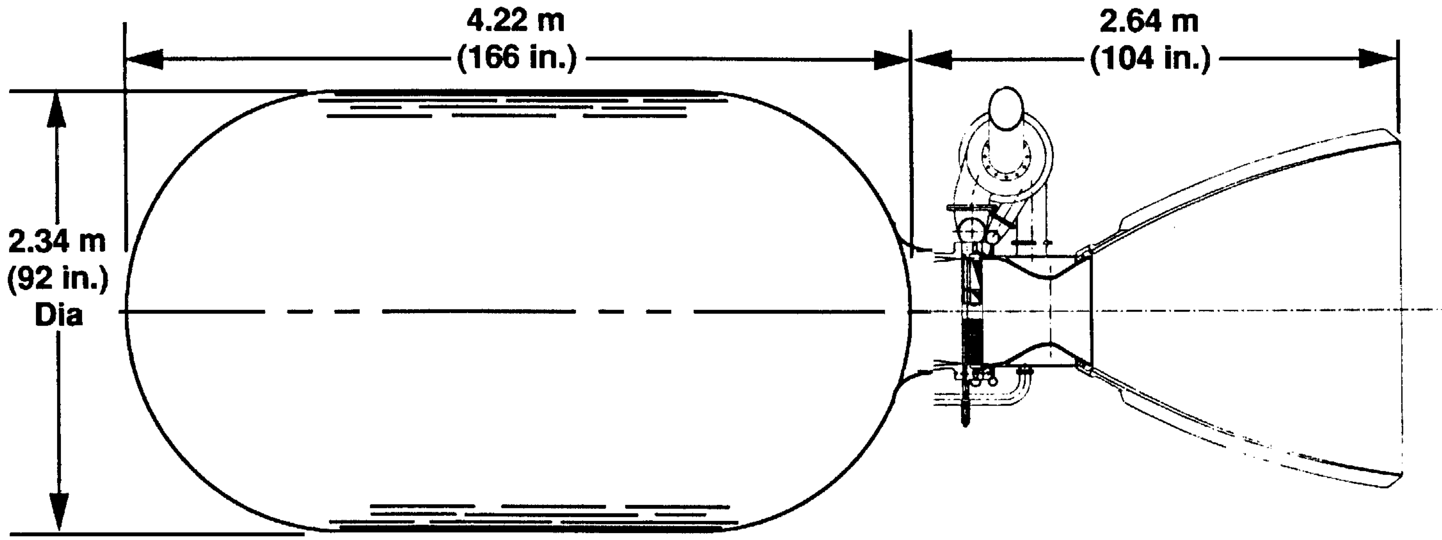
PHASE III

LARGE SUBSCALE

DEMONSTRATION PLAN

PHASE III LARGE SUBSCALE DEMONSTRATION

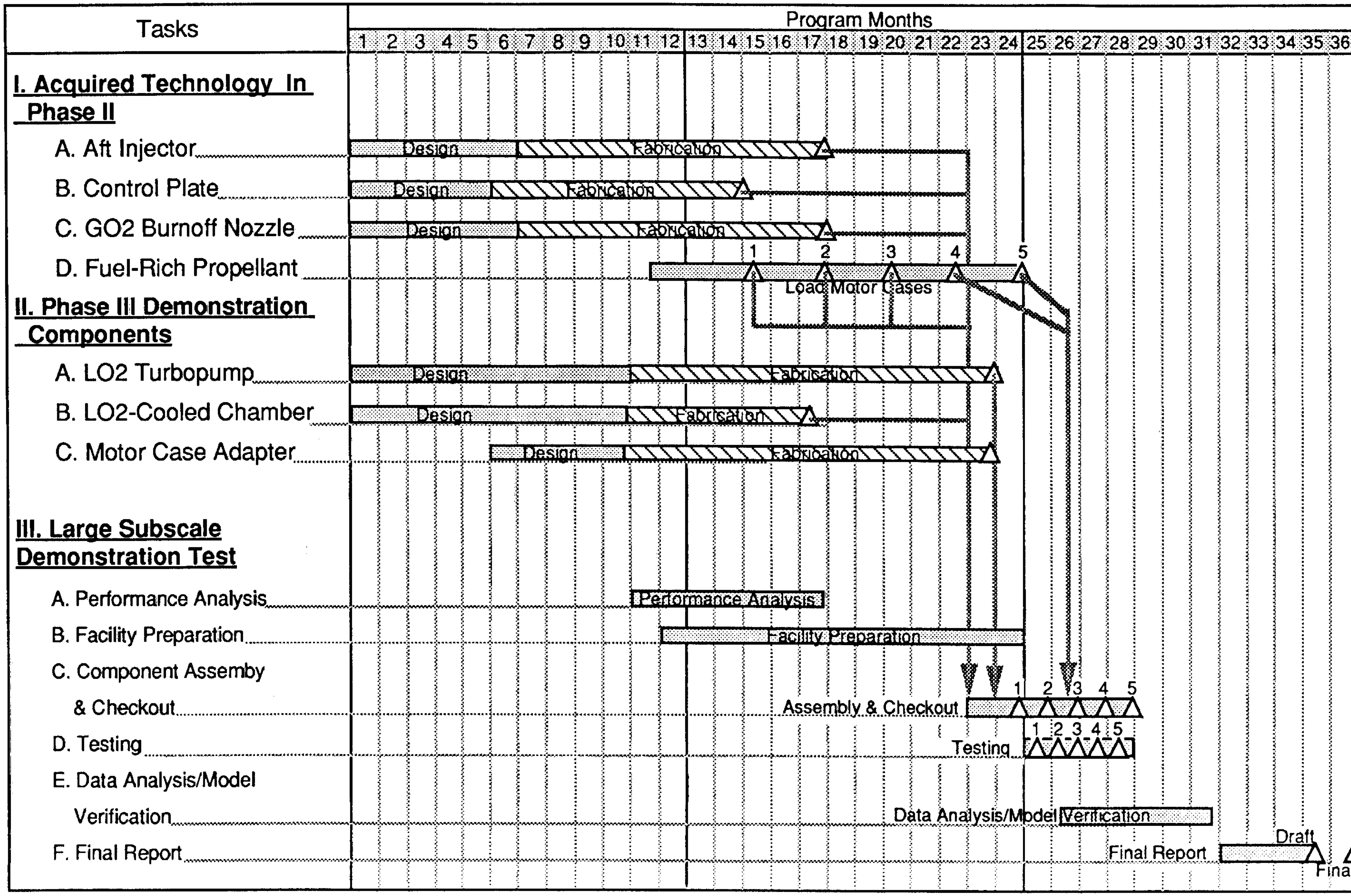
Demonstration Motor Configuration



Thrust (vac):	1.79 MN	(400,000 lbf)
Chamber Pressure:	11.72 MPa	(1,700 psia)
Specific Impulse (vac):		300 sec
Combustion Pressure (Solid):	13.09 MPa	(1,900 psia)
Total Flow Rate:	604.6 kg/sec	(1,333 lb/sec)
Engine Weight:	1270 kg	(2,600 lbm)
Solid Propellant Weight:	29948 kg	(55,000 lbm)

PHASE III - LARGE SUBSCALE DEMONSTRATION

PROGRAM SCHEDULE

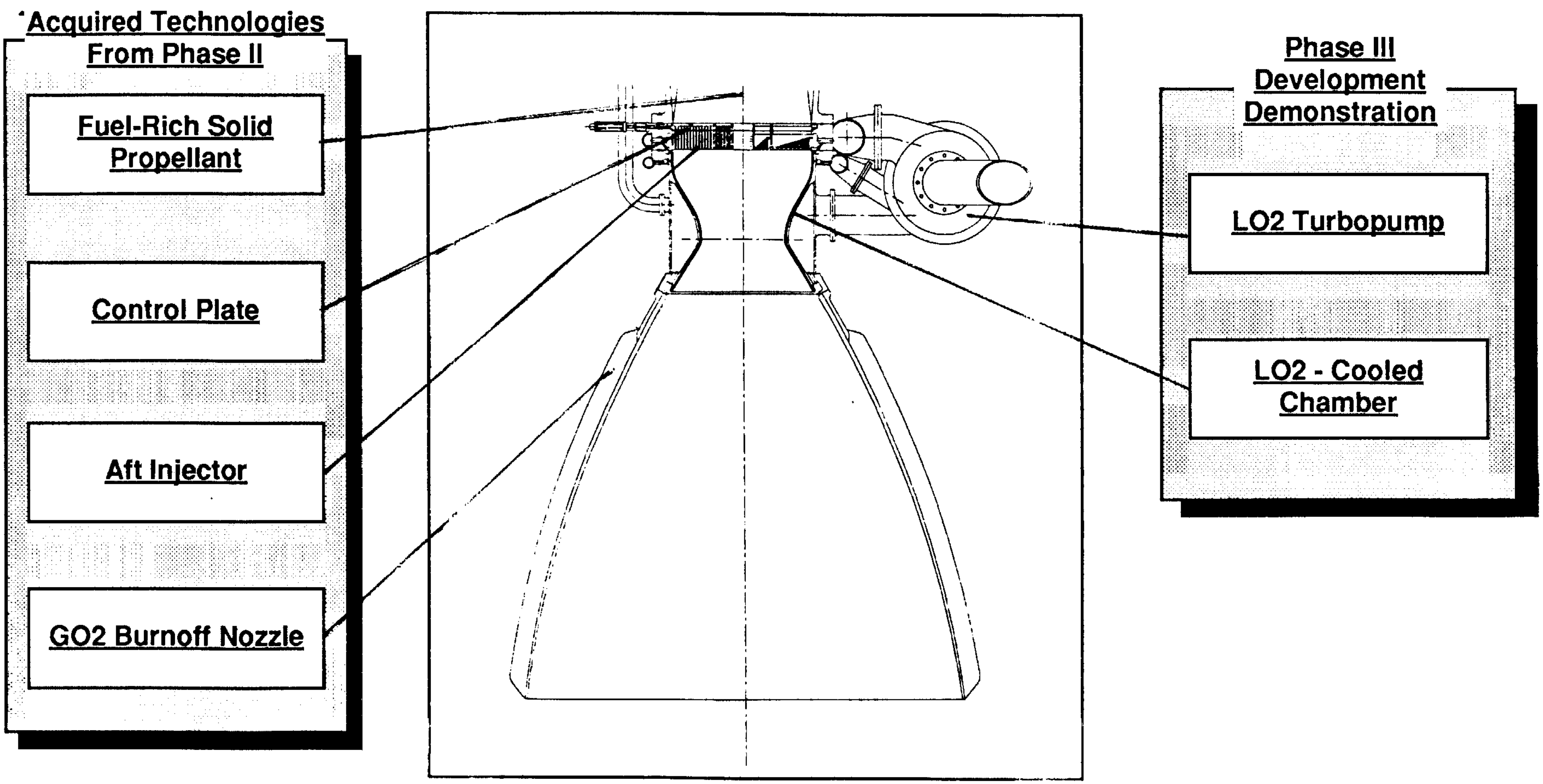


Technologies Will Be Demonstrated

For Phase III we will build large scale hardware to demonstrate our Hybrid design more closely than we did in Phase II. For this purpose we will develop a few components in Phase III but whose technology has been demonstrated previously in other programs. These are shown below along with the technology acquired in Phase II.

PHASE III - LARGE SUBSCALE DEMONSTRATION

Acquired Technologies Will Be Demonstrated



PHASE III - LARGE SUBSCALE DEMONSTRATION

A Typical Test Plan

Test No.	Duration	MR	Pc	Purpose
1	36 sec	Nominal	Nominal	Checkout Test. Verify Solid Ignition, Gas/Liquid Ignition, Performance, Stability, TPA Operation, LO2 Chamber, GO2 Burnoff, Start & Shutdown Transients.
2	36 sec	Controlled	Varying	Throttling Test: Demonstrate Throttling Operation, Verify Operation of Control Plate, LO2 Valve Control, MR Control, Performance, Stability
3	36 sec	Nominal	Nominal	Abort Test: Demonstrate Operation of the Control Plate In An Abort Mode, Verify Performance, Stability Shutdown Transients, solid Propellant Extinguihment
4	36 sec	Controlled	Varying	Combined Test: Steady State, Throttling, Full Duration, Shutdown.
5	36 sec	Nominal	Nominal	Contingency Test: Steady State, Throttling, Full Duration, Shutdown.

FACILITIES

Applicable Test Facilities Evaluated

A number of test facilities for the large subscale demonstration tests (F = 400 K lbf) were evaluated and those most applicable are shown in the table. They all satisfied the thrust, LO2 capacity and on-line availability requirements. Test stand modification costs were the deciding factors. The Aerojet, AFAL and the Stennis facilities were designed to conduct component testing thus will require modifications for HRB testing. NASA-MSFC, on the other hand, will be designed to accept the HRB and the scheduled availability is acceptable. Data acquisition capability is satisfactory for all the facilities.

PHASE III LARGE SUBSCALE DEMONSTRATION

Applicable Test Facilities Evaluated*

Facility	Test Stand	Thrust	LO2 Capacity	On Line	Modification Costs	Comment
1. Air Force Astronautics Lab. Edwards AFB.	2A	2.58 MN (580 K lbf)	7.57 m^3 (2,000 gal)	1991	Medium	• Designed for Component Development • Modifications Required for HRB
RECOMMENDED 2. NASA- Marshall Space Flight Center Huntsville, AL	116	3.11 MN (700 K)	11.36 m^3 (3,000 gal)	1992	Low	• Designed for HRB
3. NASA- John Stennis Space Center Mississippi	CTF	3.34 MN (750 K)	15.14 m^3 (4,000 gal)	1993	Medium	• Designed for Component Development • Modifications Required for HRB
4. Aerojet TechSystems Sacramento, CA	E-5	3.34 MN (750 K)	15.14 m^3 (4,000 gal)	1995	High	• Modifications Required for HRB

*Assumed Thrust Requirement of 1.79 MN (400K lbf)

PHASE III - LARGE SUBSCALE DEMONSTRATION

Overall Phase III ROM Costs

Tasks	ROM Costs
Component Design • Aft Injector • Control Plate • GO2 Burnoff Nozzle • LO2 Turbopump • Test Equiptment	$5.86 M
Component Manufacturing • Load Solid Grain • Aft Injector • Control Plate • GO2 Burnoff Nozzle • LO2 Turbopump • Test Equiptment	$17.21 M
Demonstration Testing (At NASA-MSFC) • Engineering Support • Data Analysis	$1.93 M
Total	$25 M

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D. W. Culver
R. L. Friedman
A. C. Kobayashi

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Aerojet Solid Propulsion
P. O. Box 15699C
Sacramento, CA 95852-1699

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Block 9 - Report with:

Aerojet TechSystems
P. O. Box 13222
Sacramento, CA 95813

AND

Martin Marietta Aerospace
P. O. Box 179
Denver, CO 80201

16. Abstract

The objective of this NASA CSTI Booster Propulsion Technology project is to develop a validated database, together with knowledge, understanding, and design methodology applicable to the development of high thrust, solid-liquid hybrid propulsion systems. The purpose of this study was to conduct a thorough screening to arrive at the most optimum solid-liquid booster hybrid propulsion system. Upon the selection of this design, the enabling technologies were to be defined and a plan to acquire this technology was determined.

This study determined that the solid/liquid staged-combustion cycle was the optimum design where the solid grain and liquid oxidizer are stored in separate tanks. The outlet of the solid gas-generatior case is mated to the upstream side of a gas/liquid (vane type) injector. The cooled vanes of the injector is fed (via turbopump) from the oxidizer tank through a control valve. This gas-liquid injector also allows thrust modulation by simply throttling the oxidizer flow. A control plate between the solid case and the aft injector will control the hot-gas flow to the thrust chamber for smooth throttling and abort control.

The enabling technologies identified include the optimization of the solid fuel grain, the aft vaned injector, the control plate and the GO2 nozzle injection sytems. A 30-month acquisition plan was determined for Phase II to demonstrate these technologies by testing with BATES and super-Bates motors. A Phase III large subscale test program was also included in the study.

17. Key Words (Suggested by Author(s))

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